



CITATION 650 SERIES PILOT TRAINING MANUAL

SECOND EDITION

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FOR TRAINING PURPOSES ONLY

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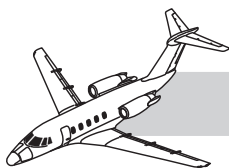
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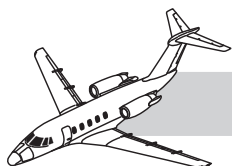


CHAPTER 1

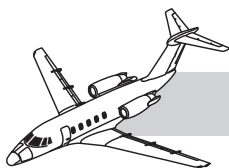
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CHAPTER 1

AIRCRAFT GENERAL



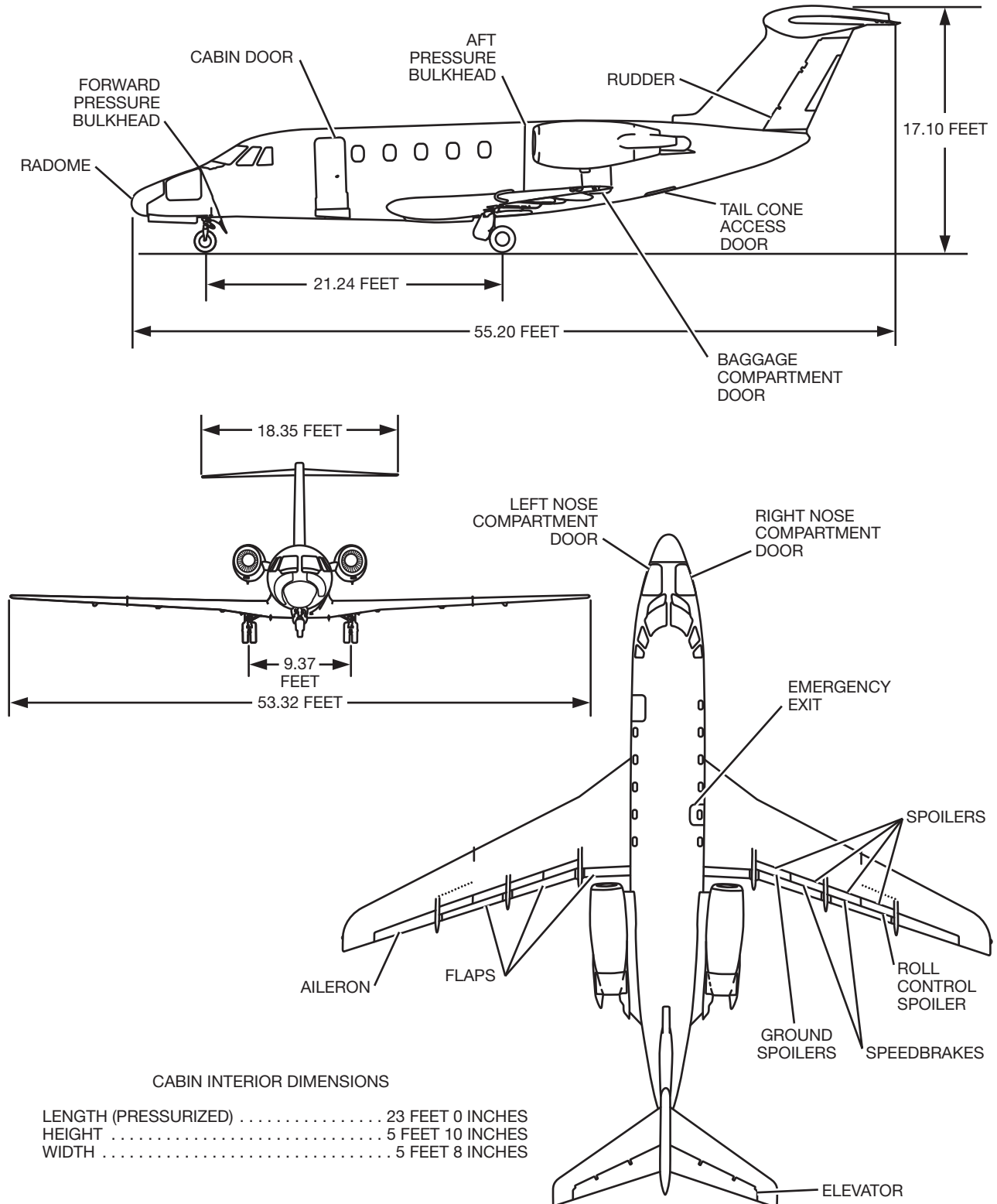
INTRODUCTION

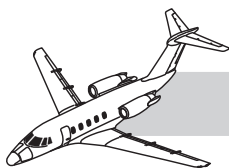
This training manual describes the main airframe and engine systems in the Citation 650 series aircraft. The information contained herein is intended only as an instructional aid. The material does not supersede, nor is it meant to substitute for, any of the manufacturer's flight or operating manuals. This chapter covers the aircraft structure and provides an overview of its systems.

GENERAL

The aircraft is a pressurized twin-engine jet aircraft with a super critical wing. It is powered by Honeywell TFE-731 engines. The aircraft may be configured to seat up to 15 people, including the crew.

The aircraft uses fail-safe construction and is certified according to FAR Part 25 airworthiness standards, including Amendment 39. High technology turbofan engines and moderately swept supercritical wings contribute to operating efficiency and performance.


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Figure 1-1. Aircraft Dimensions



The aircraft operation requires one pilot and one copilot. The pilot-in-command must have a Model 650 type rating and meet FAR 61.58 requirements for two-pilot operation. The copilot must have a multi-engine rating and meet FAR 61.55 requirements.

DIMENSIONS

Figure 1-1 shows the aircraft dimensions. Figure 1-2 shows the taxi turning distance and radii, and engine inlet and exhaust hazard areas.

STRUCTURES

NOSE SECTION

The unpressurized nose section houses various avionics components, as well as the oxygen bottle, nosewheel steering accumulator, pneumatic bottles for landing gear and brakes, nose gear wheel well, nosewheel accumulator gauge and associated dump valve, and alcohol reservoir (Citation III/VI). The nose section components are accessed through the left and right nose equipment access doors (Figure 1-3). The doors are secured by quick-release latches and key locks.

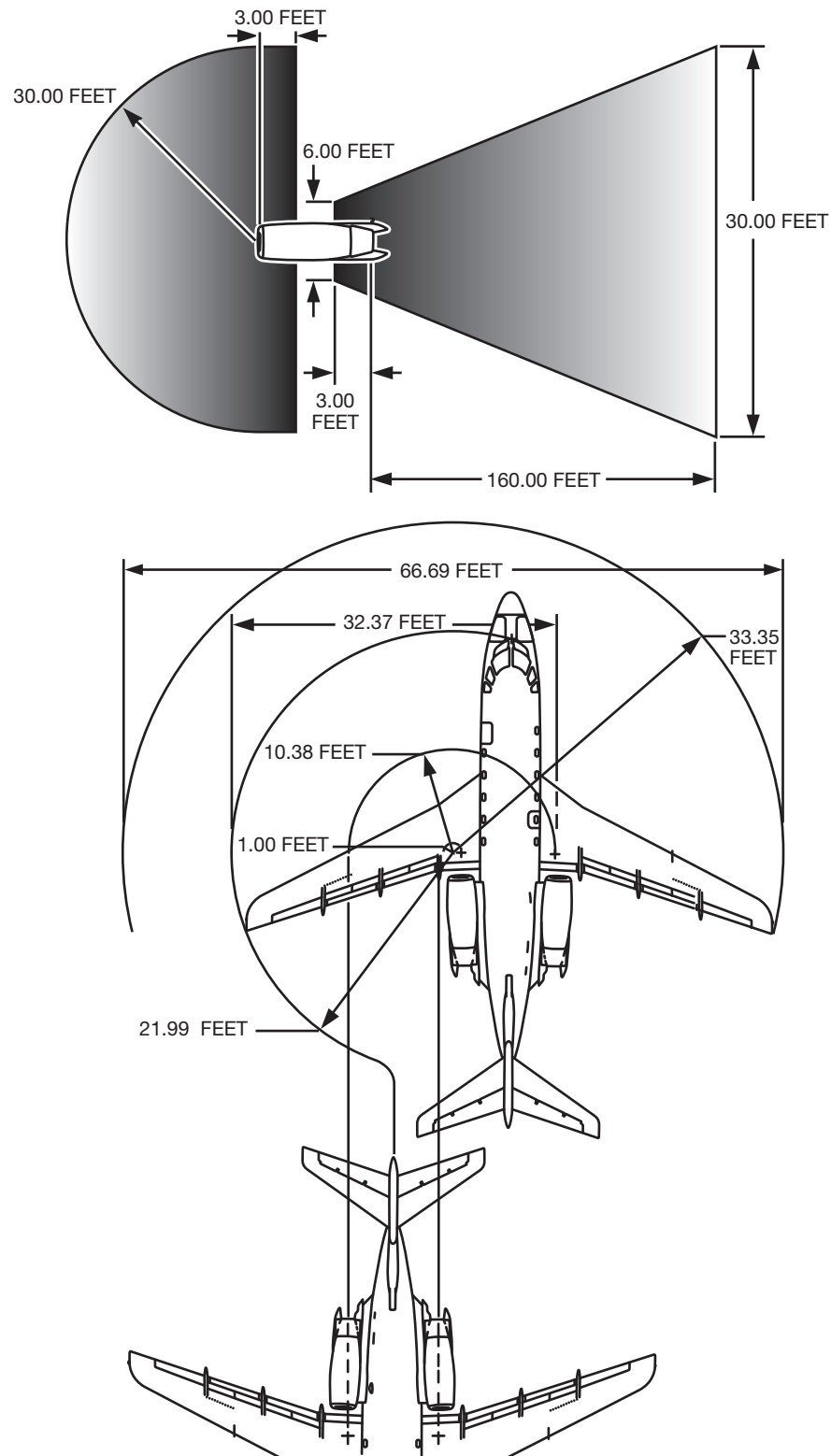
Depending on the aircraft, the ACC DOOR UNLOCKED annunciator can monitor the left nose avionics door, the single-point pressure refueling door, the tailcone baggage door, and the toilet service access door. Aircraft modified by SB 650-52-19 have a mechanical safety latch on the right nose door for additional safety. A removable radome provides access to the radar antenna.

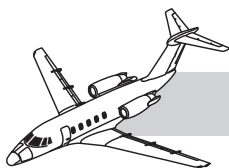
CABIN

The pressurized center section includes the passenger cabin. All interiors are custom-designed and vary greatly.



Figure 1-3. Nose Section


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Figure 1-2. Taxi Turning Radii and Engine Hazard Areas



The cabin extends from the flight compartment divider to the aft pressure bulkhead in the toilet area. A typical interior arrangement consists of six fully adjustable passenger seats. The cabin has dropout constant-flow oxygen masks for emergency use. The cabin overhead panels have individual air outlets and seat lighting for passenger comfort. Indirect lighting by fluorescent bulbs is controlled by a switch near the cabin entrance.

EMERGENCY EXIT

The emergency exit hatch is on the right side of the aircraft over the wing (Figure 1-4). The plug-type hatch opens inward. A locking pin may be inserted to prevent unauthorized entry. The pin must be removed before flight. The emergency exit is not connected to the door warning circuit.



SINGLE-POINT PRESSURE REFUELING DOOR

The single-point pressure refueling door is on the right side of the fuselage aft of the right wing (Figure 1-5). The door is hinged on the bottom and secured by quick-release fasteners and a key lock.



TOILET

The aircraft has a toilet in the aft cabin area. Depending on the model, the toilet is serviced either from inside the aircraft or through an external outlet on the lower left of the fuselage below the left engine (Figure 1-6).

FLIGHT COMPARTMENT

The flight compartment has two adjustable seats with seatbelts and shoulder harnesses, as well as dual controls, including control yokes, brakes, and rudder pedals at each crew seat (Figure 1-7).

Figure 1-4. Emergency Exit



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Figure 1-5. Single-Point Pressure Refueling Door



Figure 1-6. Toilet Access Door



Figure 1-7. Typical Instrument Panel

**Figure 1-8. Wings**

WINGS

The wings are a fail-safe design with bonded and riveted construction (Figure 1-8). The wings have a moderate sweep and use an advanced supercritical wing section. The design is optimized for low aerodynamic drag, high internal volume for structure and fuel, and favorable approach and landing characteristics. Each wing incorporates ailerons, spoilers, flaps, stall strips and fence, 11 vortex generators, fuel storage, and main landing gear. Anti-ice protection is provided by engine bleed air and an electrically heated wing cuff.

AFT BAGGAGE COMPARTMENT

Aft of the fuselage fuel tank is an electrically heated unpressurized baggage compartment with access on the left side of the aircraft (Figure 1-9). The door has integral stairs and interior lighting. Compartment lighting is provided (Figure 1-10). The door is secured by two quick-release latches and a key.

TAILCONE COMPARTMENT

The tailcone compartment is in the aft fuselage and houses various aircraft systems equipment. These systems include (Figure 1-11):

- Environmental components
- Fuel computers
- Auxiliary Power Unit (APU)
- Hydraulic components

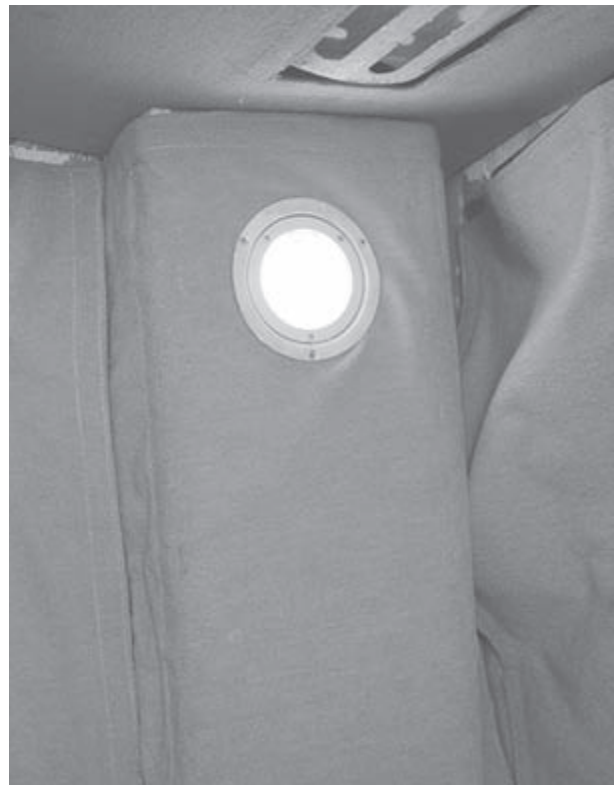
**Figure 1-9. Baggage Compartment Door****Figure 1-10. Baggage Compartment Lighting**



Figure 1-11. Tailcone Compartment

The tailcone compartment is accessed through a door on the bottom of the fuselage. The door is hinged on the forward edge and secured by three quick-release latches and a key lock. Interior lighting is provided and the door may be monitored by the ACC DOOR UNLOCKED annunciator.

EMPENNAGE

The empennage is a T-configuration (Figure 1-12). The leading edge of the horizontal stabilizer incorporates a bird splitter and can withstand FAR 25 bird impact loads.



Figure 1-12. Empennage

The horizontal stabilizer is a one-piece swept design. Pitch trim is accomplished by varying the incidence of the horizontal stabilizer.

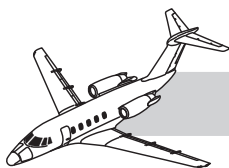
The leading edge is anti-iced by electrical power. A stick shaker ensures adequate stall warning in all configurations.

The vertical stabilizer and rudder are a conventional design. Rudder operation is manual, but a rudder bias system connected to the rudder minimizes yaw associated with asymmetric thrust situations.

SYSTEMS

ELECTRICAL POWER SYSTEM

Two starter-generators supply power to the aircraft DC buses. Engine starting and secondary DC power is available from batteries, external power, or an optional onboard APU. Two static inverters provide avionics AC power.



LIGHTING SYSTEM

The aircraft has the following lights:

- Standard navigation
- Anticollision
- Recognition
- Landing
- Taxi
- Wing inspection
- Tail flood light (on some models)

Interior lighting is provided for the cockpit and cabin, as is lighting for the baggage and tailcone compartments.

A cabin entry switch is on the forward interior side of the main entrance door frame. Emergency lighting is provided by two NiCad batteries for general cabin illumination, emergency exit illumination and identification, and evacuation path and ground illumination.

MASTER WARNING SYSTEM

System conditions or faults are indicated automatically through amber and red annunciators and a MASTER WARNING RESET switchlight.

The MASTER WARNING RESET switchlight can be reset when flashing and illuminates with selected system annunciators. Various aural warnings also sound to indicate certain conditions or system malfunctions.

FUEL SYSTEM

The aircraft fuel system uses integral wing tanks and a bladder fuselage tank. Fuselage fuel can be transferred to the wing tanks as wing tank fuel is depleted. Manual control and monitoring of the fuel system is available and fuel transfer between the wings enables fuel availability to either engine. Single-point pressure refueling is available through a port aft of the right wing root; overwing refueling can be accomplished as well.

APU SYSTEM

The aircraft can be outfitted with a number of different ground only or ground/air APUs. APUs provide electrical power to the DC system and bleed air to the environmental control units. If a hydraulic pump is installed, the APU can provide pressure to the main hydraulic system.

ENGINES

The aircraft are powered by two Honeywell TFE-731 turbofan engines producing either 3,650 or 4,080 pounds of thrust, depending on the model.

The engines are two-spooled, medium or high bypass with a geared fan and are modularized for ease of maintenance. Bypass air cools the turbine section and reduces engine-generated noise, while the reverse-flow annular combustion chamber reduces engine length and weight.

Efficient fuel scheduling is accomplished by an electrohydraulic fuel control and either an electronic fuel computer or a digital electronic fuel computer. Either fuel computer automatically maintains an economical and precise fuel schedule throughout the entire spectrum of atmospheric conditions and thrust requirements.

High and low pressure bleed air is extracted from the compressors for pressurization, air conditioning, anti-icing, and other systems. Thrust reverser levers are piggyback mounted controls on the throttles.

FIRE PROTECTION

Fire and overheat protection is provided in the engine nacelles. The fire protection system includes a detector/sensor, detection control unit, and LH or RH ENG FIRE PUSH switchlight for each engine.

Fire-extinguishing capability is provided by two fire-extinguishing containers, which can


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direct the extinguishing agent of both containers to either engine.

Two hand-held portable fire extinguishers, one under the copilot seat and the other in the cabin, provide fire protection inside the aircraft.

PNEUMATICS

Engine compressor bleed air is extracted from the low or high pressure compressor cases for air conditioning and pressurization. Only high pressure bleed air is used for all other bleed air systems.

ICE AND RAIN PROTECTION

The aircraft is approved for flight into known icing conditions when the required equipment is operational.

The ice protection systems prevent or dispose of ice or rain on critical areas of the aircraft. The anti-ice energy sources and their protected components are:

- Engine compressor bleed air—Nacelle air inlet, wing leading edges, and windshield.
- Electric heat—Horizontal stabilizer, P_{T2} T_{T2} probes, generator air inlets, rudder bias heater, wing root fairing, angle-of-attack sensors, pitot tubes, static ports, ram-air temperature probe (and windshields on Citation VII only).
- Fluid—Backup alcohol anti-icing of the pilot windshield (Citation III/VI only).

AIR CONDITIONING

The air conditioning system uses engine bleed air for heating, cooling, and pressurization.

The environmental control panel switches allow selection of either one or both engines to provide a controlled volume of high or low pressure bleed air to either or both environmental control units (ECUs), also referred to as pneumatic air conditioning units (PACs).

The left engine furnishes bleed air to the cockpit ECU. The right engine furnishes bleed air to the cabin ECU.

PRESSURIZATION

The pressurization control system regulates the amount of air allowed to escape the pressurized portion of the fuselage, thereby controlling cabin pressure to the desired altitude.

Maximum differential pressure is normally 9.3 psi, which equates to a cabin altitude of 8,000 feet at 51,000 feet. The cabin pressurization system uses conditioned air to maintain a lower cabin altitude than aircraft altitude.

Cabin pressurization is accomplished by maintaining a relatively constant air flow into the cabin and controlling air outflow through two outflow valves on the aft pressure bulkhead.

Outflow air and cabin altitude are controlled through the cabin pressurization controller or a manual backup controller.

HYDRAULIC SYSTEM

Variable displacement engine-driven pumps supply pressure for operation of the spoilers, aileron boost, landing gear, wheel brakes, nosewheel steering, and thrust reversers.

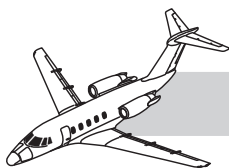
An electric auxiliary pump supplies pressure for the roll control spoilers and the wheel brakes if the main hydraulic system fails.

Pneumatic backup systems are available for landing gear extension and wheel brakes. Nosewheel steering is electrically enabled, manually controlled, and hydraulically actuated.

LANDING GEAR AND BRAKES

The landing gear is electronically controlled and hydraulically actuated. The gear is held in the up position by mechanical uplocks and is mechanically locked down when extended.

Braking is provided by a Hydro Air Mark II antiskid brake system with individual wheel



control. The system features touchdown protection and locked wheel protection.

Power braking with antiskid protection is provided with emergency pneumatic braking as a backup. Automatic braking occurs after take-off during gear retraction to stop the rotation of the main wheels prior to their entering the wheel wells.

FLIGHT CONTROLS

Primary roll, pitch, and yaw control is provided through typical cable-operated controls. Rudder and aileron trim is accomplished through manual controls.

Pitch trim is accomplished by electronically changing the incidence of the stabilizer by a primary or secondary trim system.

Roll control is aided by an electrically controlled, hydraulically actuated aileron boost system and mechanical/hydraulic roll spoilers. Four hydraulically operated spoilers are on top of each wing. Two are used as speedbrakes, and all four, including the roll-control spoilers, are used after landing and during emergency descent.

AVIONICS

Specific avionics systems vary with aircraft serial number and customer preference. Many optional avionics items are available. Early serial numbers were provided with a mechanical flight instrument system with an Electronic Flight Instrument System (EFIS) option. Later serial numbers were provided with a complete, digital integrated flight control system. Equipment may include:

- Weather radar
- Altitude reporting transponder
- Autopilot/flight director
- EFIS with optional MFD
- TCAS
- EGPWS

Communication is provided by two VHF transceivers and an optional HF transceiver. Navigation equipment includes digitally tuned ADF, DME and two VOR/LOC glideslope and marker beacon receivers.

MISCELLANEOUS

The aircraft was originally equipped with a 49-cubic-foot standard oxygen bottle. In later models a 76-cubic-foot oxygen bottle became standard, with the option for those aircraft equipped with the 49-cubic-foot bottle to upgrade to the larger bottle.

The oxygen cylinder is on the lower left side of the nose. The oxygen system supplies the cockpit through quick-donning pressure-demand masks.

Oxygen is supplied to the cabin through dropout constant flow masks, which are deployed automatically at approximately 13,500 feet cabin altitude.

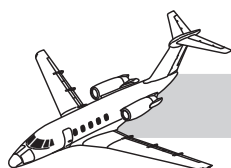
PUBLICATIONS

The following publications must be immediately available to the flight crew:

- FAA-approved *Airplane Flight Manual (AFM)*—Includes limitations, takeoff and landing data, and weight and balance data. Information in the *AFM* always takes precedence over any other publication.
- Appropriate FAA-approved Citation 650 *Abbreviated Checklists*—Normal, Emergency, and Abnormal Procedures include abbreviated operating procedures and abbreviated performance data. If any doubt exists or if the conditions are not covered by the checklist, the *AFM* must be consulted.

Other publications:

- *Operating Manual*
- FAA-approved Weight and Balance Manual

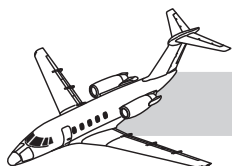


CHAPTER 2

ELECTRICAL POWER SYSTEMS

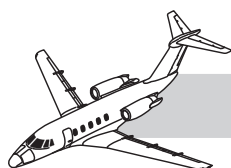
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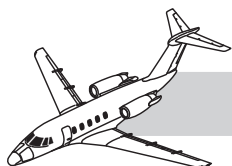
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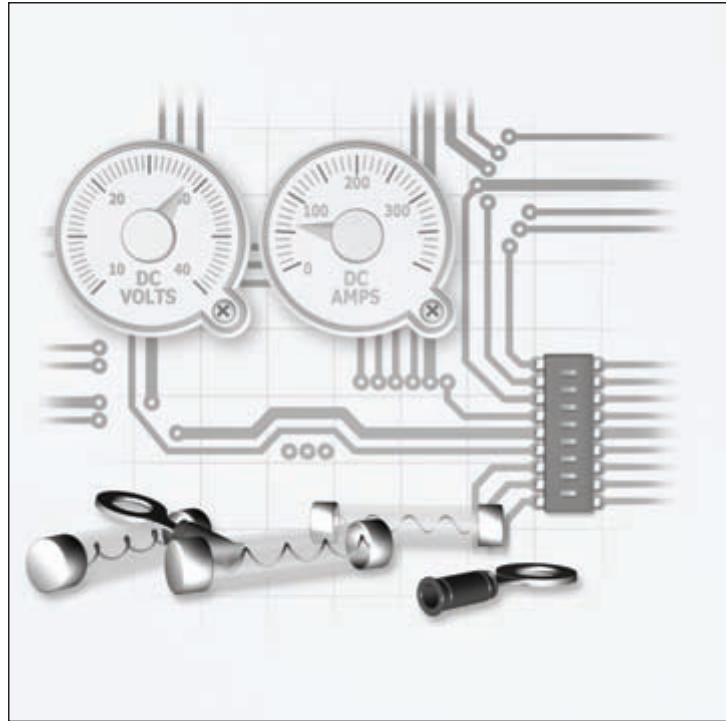
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CHAPTER 2

ELECTRICAL POWER SYSTEMS



INTRODUCTION

This chapter provides a description of the electrical power system used on the Citation 650 series aircraft. Included is information on the DC and AC systems. The DC system consists of storage, generation, distribution, and system monitoring. The AC system consists of generation, distribution, and system monitoring. Provision is also made for a limited supply of power during emergency conditions in flight and for connection of an external power unit while on the ground.

GENERAL

The Citation 650 series aircraft incorporates a DC and an AC electrical system. DC electrical power is required for operation and control of main airplane systems such as hydraulics, environmental, and anti-ice. AC electrical power is required for various avionics systems such as navigation, communication, autopilot, and

radar. AC power is also required for various anti-ice systems. DC electrical power is provided by two starter-generators mounted on the engines and connected in parallel to a common bus system for equal load-sharing. Secondary and backup DC power sources are provided by nickel-cadmium batteries and an



optional onboard auxiliary power unit (APU). Provision for connecting an external power supply unit (EPU) when on the ground also is incorporated.

The primary sources of AC electrical power required for avionics are two solid-state inverters that convert main DC electrical power into 115-volt and 26-volt AC power. AC electrical power for the anti-ice is provided by engine driven alternators.

All electrical buses, wiring, and equipment are protected by current limiters and circuit breakers. Backup and emergency power supplies with associated buses and circuits are incorporated to provide adequate electrical power for both AC and DC essential equipment during emergency operations. Load-shedding procedures enable the crew to reduce electrical loads by removing power from nonessential equipment but maintain essential electrical power during emergency situations caused by a loss of primary power. A DC voltmeter, ammeter gauges, annunciator lights, and master warning lights provide monitoring capability for the electrical system.

DC POWER

STARTER-GENERATORS

The principal sources of power are the engine-driven starter-generators. Rated at 365 amps (50% overload for five minutes) and 30 volts, that are regulated to 28.5 volts output. During engine start, the generators serve as starter motors. After termination of the start sequence, it reverts to its generator function. It is limited to three starts in any 30-minute period with one minute between starts.

The generator is limited to 365 amps for takeoff and 300 amps for all other ground operations. A single generator is capable of supporting the entire electrical system requirements.

BATTERIES

The standard battery configuration consists of two nickel-cadmium batteries rated at 24 volts

and 22, 40, or 44 ampere-hours. Lead acid batteries are also available for use. The batteries are connected in parallel, are located under the baggage compartment flooring and are provided with quick disconnects (Figure 2-1).



Figure 2-1. Battery Location

The batteries are a secondary source of DC power that are used to provide power for the starting sequence and to provide power to the hot battery and emergency buses in the event of a dual generator failure.

The batteries are susceptible to, and must be protected from, overheating due to excessive charging. Therefore, they are limited in use for engine starting to three starts per hour.

An engine start with the assistance of an operating generator is considered one-third of a battery start. External power unit (EPU) starts do not involve the batteries. Refer to Table 2-1 for battery and starter cycle limitations.

The batteries connect to the hot battery bus. A battery disconnect relay is installed between each battery and its ground to provide for electrically disconnecting one or both batteries during certain conditions.

A BATT DISC switch (Figure 2-2) is installed in the cockpit of dual-battery airplanes. This switch opens one battery disconnect relay at a time in case of battery overheating. Care should be taken to keep this switch in the NORM position to maintain an equal charge on the batteries.



Table 2-1. BATTERY AND STARTER CYCLE LIMITATIONS

TYPE LIMIT	LIMITATION
STARTER (1)	THREE ENGINE STARTS PER 30 MINUTES. THREE CYCLES OF OPERATION WITH ONE-MINUTE REST PERIOD BETWEEN CYCLES ARE PERMITTED.
	THE STARTER-GENERATOR IS LIMITED TO 365 AMPERES FOR TAKEOFF AND 300 AMPERES FOR ALL OTHER GROUND OPERATIONS.
BATTERY (2, 5)	THREE ENGINE STARTS PER HOUR (REFER TO NOTES 3 AND 4).

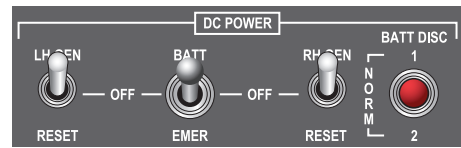
NOTES:

1. THIS LIMITATION IS INDEPENDENT OF STARTER POWER SOURCE; I.E., BATTERY, GENERATOR-ASSISTED START, EXTERNAL POWER UNIT, OR AUXILIARY POWER UNIT.
2. IF BATTERY LIMITATION IS EXCEEDED, A DEEP CYCLE INCLUDING A CAPACITY CHECK MUST BE ACCOMPLISHED TO DETECT POSSIBLE CELL DAMAGE.
3. THREE GENERATOR-ASSISTED STARTS ARE EQUAL TO ONE BATTERY START.
4. IF EXTERNAL POWER UNIT IS USED FOR START, NO BATTERY CYCLE IS COUNTED.
5. USE OF AN EXTERNAL POWER SOURCE WITH VOLTAGE IN EXCESS OF 28.5 VDC OR CURRENT IN EXCESS OF 2,000 AMPERES MAY DAMAGE THE STARTER.

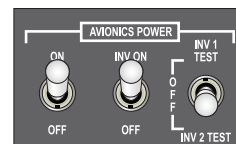
A red guarded switch labeled “STARTER DIS-ABLE” is located in the baggage compartment (Figure 2-3). When the switch is turned on, the battery disconnect relays are energized open, and the external power relay opens. This removes all power from the hot battery bus (provided a generator is not on line). This switch is not used to disconnect the batteries for an extended period because battery power is used to hold the relays open and results in battery discharge.

EXTERNAL POWER

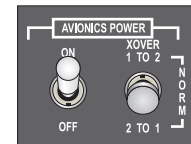
An external power unit may be connected to the airplane DC system through a receptacle on the bottom left side of the fuselage, forward of the baggage compartment door (Figure 2-4). External power is connected to the hot battery bus and routed to all buses with the battery switch in BATT, or to the emergency bus with the battery switch in EMER. Normally, the battery disconnect relays are energized open when an external power unit is connected and the batteries are isolated from the electrical system. In this configuration, the batteries are not charged by the EPU and are not involved in engine starts. Before connecting an EPU, its voltage should be regulated to 28.5 volts and the amperage to no more than 2,000 amps.



BATTERY AND GENERATOR SWITCHES



**SNs 0001—0178,
0200—0202,
0207—0241**



**SNs 0179—0199,
0203—0206,
7001—7119**

Figure 2-2. AC/DC Power Switches



Figure 2-3. Baggage Compartment Switches



Figure 2-4. External Power Receptacle

When external power is applied to the airplane, the battery disconnect relays are energized, which opens the battery ground circuits and disconnects the batteries from the hot battery bus. External power automatically disconnects by means of an external power relay in the event that a generator switch is placed to the GEN position after an engine has been started.

A switch located near the upper forward baggage door sill (Figure 2-3) closes the battery disconnect relays and allows the EPU to charge the batteries. It is labeled “BATT CHARGE” and is covered by a black guard. With the switch in the CHRG position, ground is removed from both battery disconnect relays, causing them to remain closed while EPU power is supplied, thus allowing the batteries to charge. The BATT CHARGE switch should be in NORM when connecting or disconnecting the EPU and during engine starting. EPU charging of the batteries is accomplished by adjusting EPU voltage to $28.5 \pm .5$ volts and connecting the unit to the airplane while the BATT CHARGE switch is in NORM. The switch is then placed to CHRG and the voltage readjusted to maintain $28.5 \pm .5$ volts. To stop charging, place the switch to NORM and close the guard.

NOTE

Do not charge the batteries with an EPU:

- If battery voltage is less than 23 volts
- If OAT exceeds 100°F (38°C)

NOTE

Both batteries must be charged simultaneously in dual-battery installations (both batteries must be connected to the airplane with the BATT DISC switch in the cockpit positioned to NORM).

An overvoltage protection system is provided during use of an external power unit. The control unit monitors the external power unit voltage and deenergizes the external power relay if the voltage exceeds 32.5 volts. External power cannot be reapplied to the airplane until the voltage is reduced below 32.5 volts and the control unit is deenergized to reset the monitor. This is accomplished by removing and then reinserting the EPU plug with the proper voltage set.

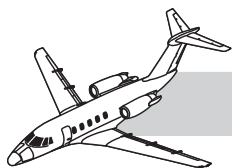
DISTRIBUTION

General

Direct current is distributed throughout the early 650 serial numbers by ten buses and in later serial numbers by 12 buses (Figures 2-5 and 2-6). Located in the main junction box in the baggage compartment are two feed DC buses, the crossfeed bus, and the hot battery bus. Located in the cockpit are the pilot and copilot circuit breaker panels which contain the circuit breakers for the two extension buses, five branch buses and the emergency bus.

The two DC feed buses (left or right) are normally powered by the respective generator and are tied together by the crossfeed bus. They may also receive power from the batteries, APU, and EPU.





CITATION 650 SERIES PILOT TRAINING MANUAL

2 ELECTRICAL POWER SYSTEMS

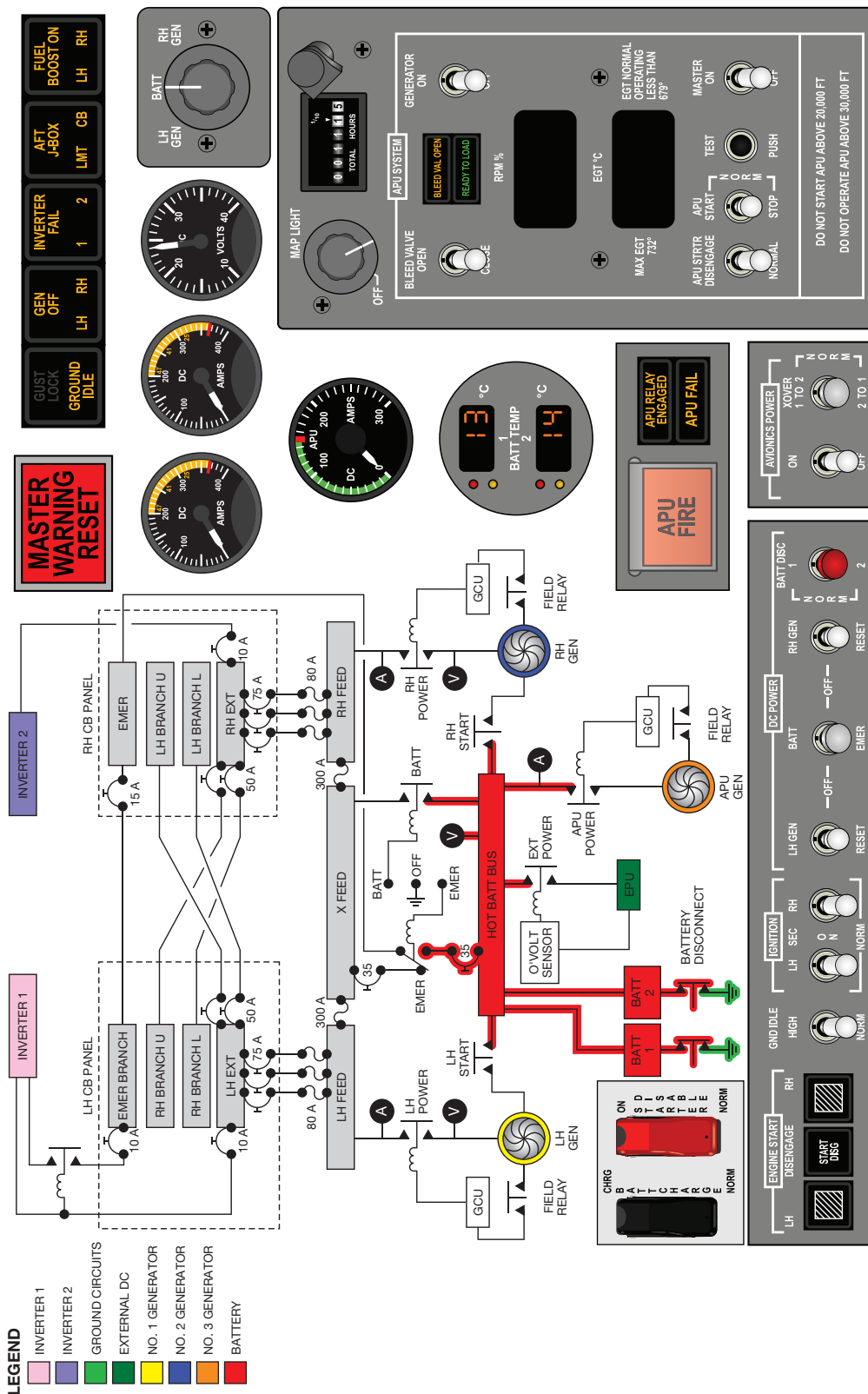


Figure 2-6. Electrical System Distribution: Later Model Serial Numbers



The batteries are connected directly to the hot battery bus. The hot battery bus can also receive power from the APU or an EPU. With the generator switches in OFF, the battery switch in OFF, and the batteries or EPU connected, only the hot battery bus is powered (Figures 2-7, 2-8, and 2-9).

Primary items that receive power directly from the hot battery bus are:

- Lights—Cabin entrance, baggage compartment, tailcone, emergency exit (battery switch—EMER)
- Auxiliary hydraulic pump
- Primary pitch power
- Primary pitch control
- Voltmeter
- Fuselage tank fill
- Ignition (start only)
- Battery disconnect switch
- Nosewheel steering accumulator dump valve

Emergency Bus

Placing the battery switch to EMER energizes the emergency relay and connects the emergency bus to the hot battery bus (Figure 2-10). The emergency bus is located in the cockpit on the right side circuit breaker panel. Items on the emergency bus are primary communications and navigation equipment necessary for flight.

Emergency bus items for SNs 0001—0178 are the following:

- Standby attitude indicator
- Air data computer No. 1
- Directional gyro No. 1 (pilot C14D gyro)
- Navigation radio No. 1/OBI (audio via headset)
- Communications radio No. 1 (audio via headset)
- Radio magnetic indicator No. 1 (EFIS-equipped airplanes only)

- Cockpit floodlights (SNs 0001—0066)
- Horizontal stabilizer trim advisory (indicator and clacker)
- Emergency branch bus circuit breaker
- Navigation transfer (SNs 0067—0178)

Emergency bus items for SNs 0200—0202 and 0207—0241 are the following:

- Standby attitude indicator
- Digital air data computer No. 1
- Communications radio No. 1
- Navigation radio No. 1/OBI
- Radio magnetic indicator No. 1
- Audio (overhead speakers) 1 and 2
- Standby Nav/Comm (Primus II only)
- Directional gyro No. 2
- Radio management unit No. 1 (Primus II only)
- Emergency branch bus circuit breaker
- Horizontal stabilizer trim advisory (indicator and clacker)

Emergency bus items for SNs 0179—0199, 0203—0206, 7001—7119 are the following:

- Standby attitude indicator (SNs 0179—0199 and 0203—0206)
- Digital air data computer No. 1
- Communications radio No. 1
- Navigation radio No. 1/OBI
- Radio magnetic indicator No. 1
- Audio 1 and 2
- Pilot altimeter
- Horizontal stabilizer trim advisory (indicator and clacker)
- Attitude and heading reference system No. 2
- Standby Nav/Comm (Primus II only)
- Radio management unit No. 1 (Primus II only)



CITATION 650 SERIES PILOT TRAINING MANUAL

2 ELECTRICAL POWER SYSTEMS

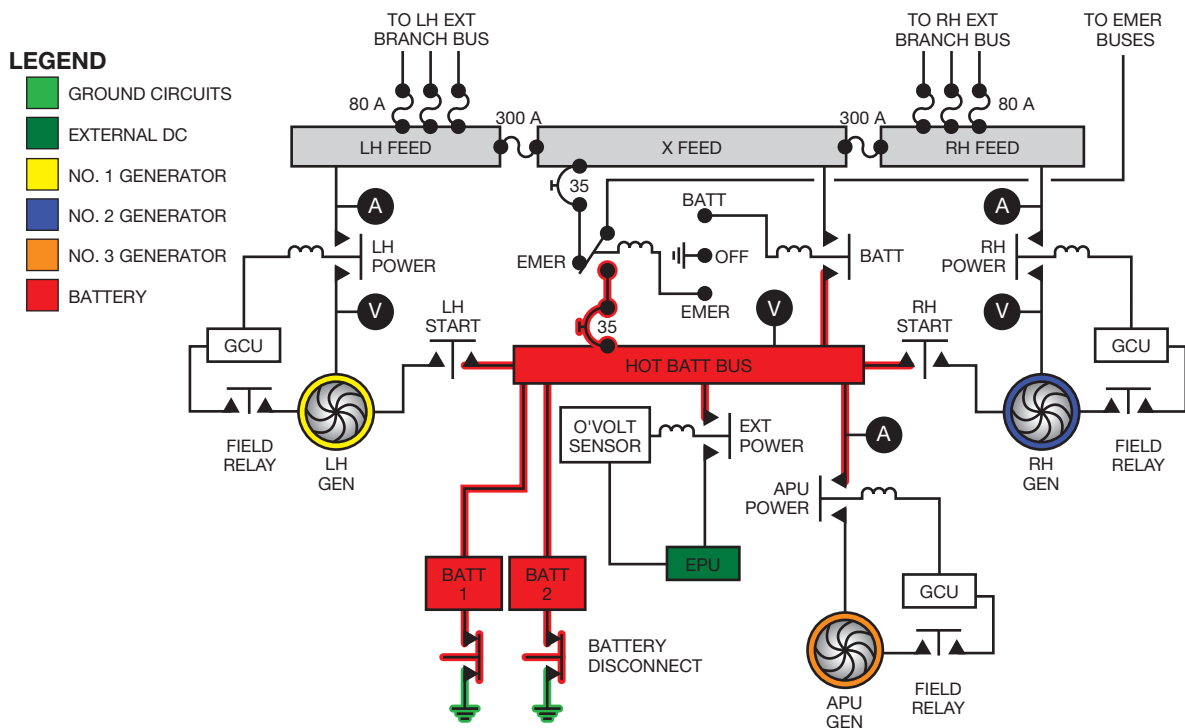


Figure 2-7. Electrical Circuit Condition—Battery Switch Off

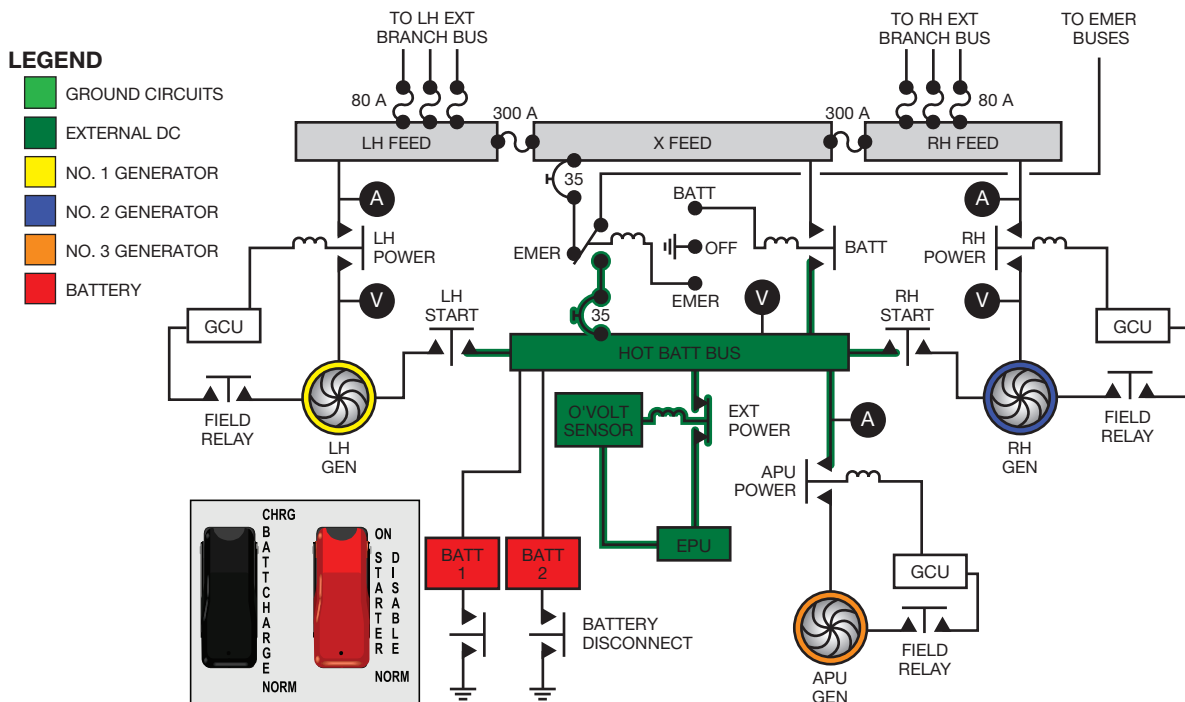


Figure 2-8. Electrical Circuit Condition—Battery Switch Off, EPU Connected



CITATION 650 SERIES PILOT TRAINING MANUAL

LEGEND

- GROUND CIRCUITS
- EXTERNAL DC
- NO. 1 GENERATOR
- NO. 2 GENERATOR
- NO. 3 GENERATOR
- BATTERY

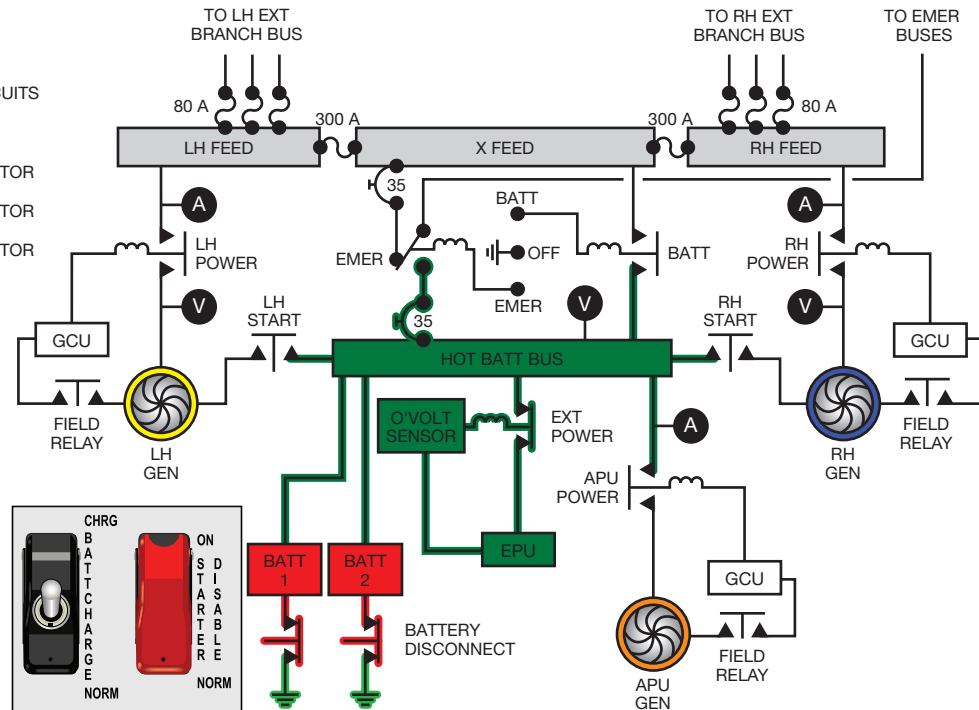


Figure 2-9. Electrical Circuit Condition—Battery Switch Off, EPU Connected, Charge Switch in CHRG

LEGEND

- GROUND CIRCUITS
- EXTERNAL DC
- NO. 1 GENERATOR
- NO. 2 GENERATOR
- NO. 3 GENERATOR
- BATTERY

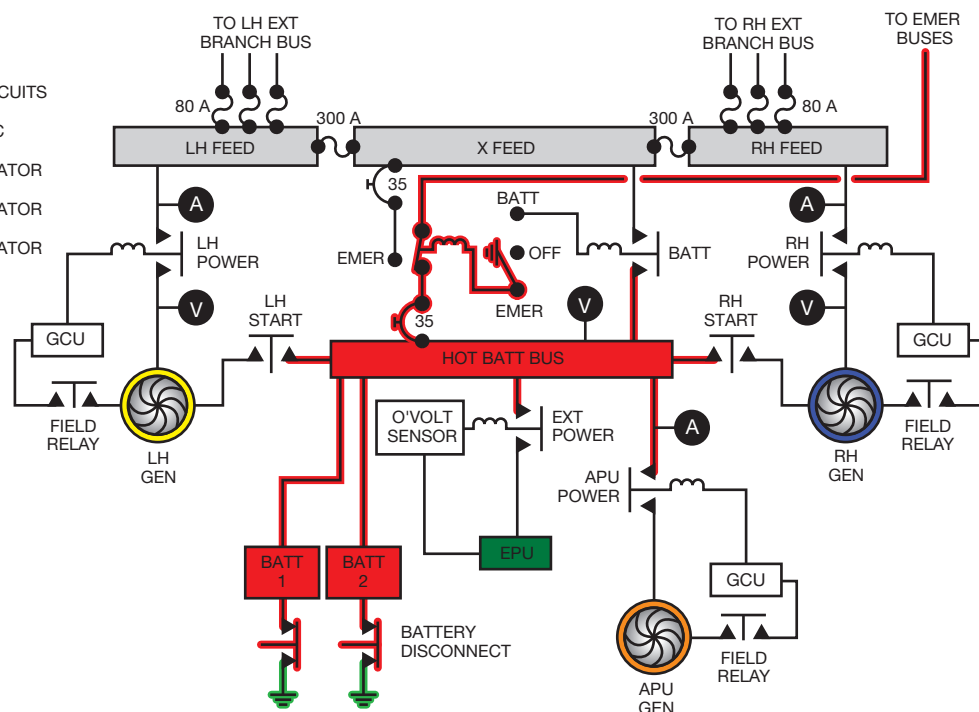
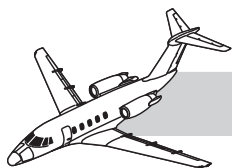


Figure 2-10. Electrical Circuit Condition—Battery Switch in EMER



Emergency Branch Bus

The emergency branch bus is connected to the emergency bus by a circuit breaker located on the copilot circuit breaker panel. The emergency branch bus is wired across the cockpit and located on the pilot circuit breaker panel. The emergency branch bus provides power to equipment necessary for monitoring the engines and operation of secondary pitch trim (backup). Emergency branch bus items are:

- Left/right fan rpm (N₁) (all aircraft)
- Left/right ITT (all aircraft)
- Secondary pitch trim (all aircraft)
- Cockpit floodlights (all aircraft except SNs 0001—0066)
- Left/right fire detection (all aircraft except SNs 0001—0151)
- Left/right firewall shutoff (all aircraft except SNs 0001—0151)
- Left pitot-static anti-ice (all aircraft except SNs 0001—0178)
- AC inverter No. 1 (SNs 0179—0199, 0203—0206, 7001—7119 only)

Crossfeed and Left and Right Feed Buses

Placing the battery switch to BATT energizes the battery relay closed and connects the crossfeed bus to the hot battery bus (Figure 2-11). With the batteries or an EPU connected, the battery switch in BATT, all current limiters intact, all circuit breakers in, and both generator switches in OFF, all buses are powered by the batteries or the EPU.

Main DC power is distributed from the hot battery bus through the battery relay to the crossfeed bus. The crossfeed bus is a bus tie that connects the left and right feed buses. The crossfeed bus also serves as the direct power source for the flaps. It also powers the emergency buses when the battery switch is not in the emergency position. Each feed bus is connected to the crossfeed bus by a 300-ampere current limiter for protection.

On SNs 0001—0173 only, the feed buses, crossfeed bus, and hot battery bus are located in the overhead of the baggage compartment in the main electrical junction box. Two auxiliary junction boxes are located in the tailcone.

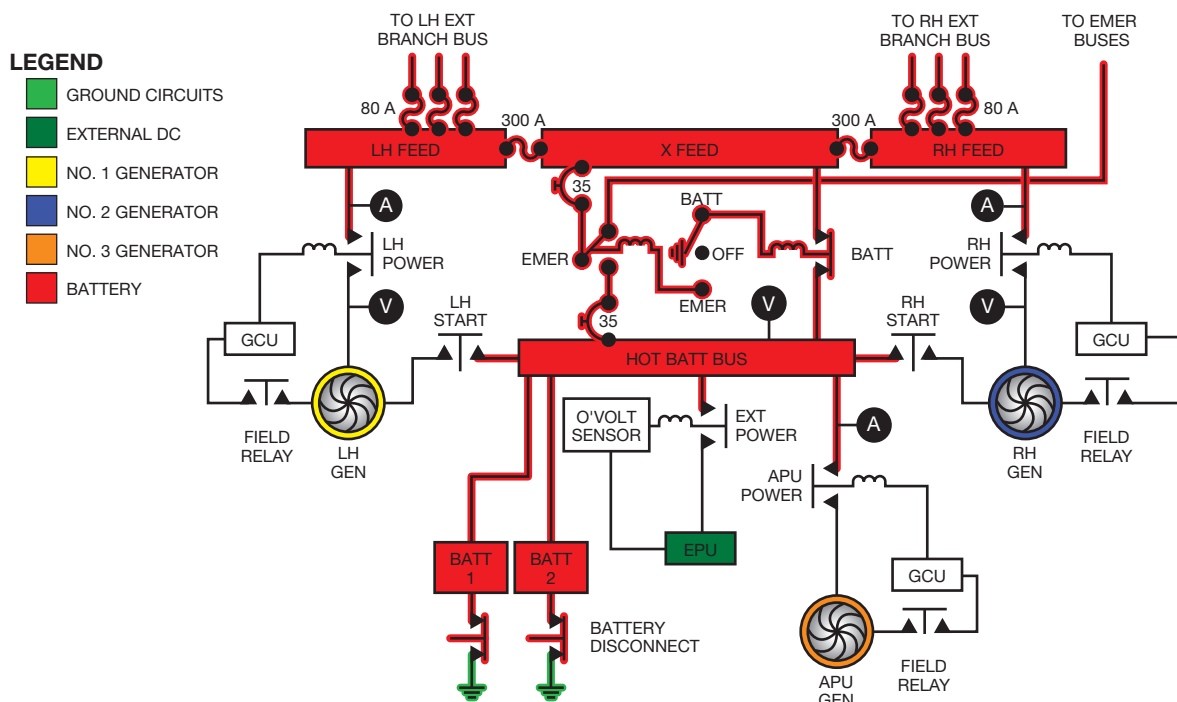


Figure 2-11. Electrical Circuit Condition—Battery Switch in BATT



These “J” boxes are connected directly to the feed buses for power distribution to anti-ice equipment. On later SNs the junction box is attached to the rear bulkhead of the baggage compartment.

Left and Right Extension Buses

DC electrical power is distributed to the cockpit via three wires in parallel from each feed bus to its respective extension bus. These wires are protected by 80-ampere current limiters on the feed buses and are connected to the extension buses by three 75-ampere circuit breakers located on each cockpit circuit breaker panel. The left extension bus is located on the pilot circuit breaker panel and the right extension bus on the copilot circuit breaker panel. The three 75-ampere circuit breakers are labeled “No 1,” “No 2,” and “No 3 LH–RH FEED” and are located on the lower rear section of each cockpit circuit breaker panel.

Left and Right Branch Buses

The extension buses are further expanded by the branch buses, which are wired across the cockpit to the opposite circuit breaker panel. The left branch buses are located on the right circuit breaker panel, and the right branch buses are located on the left circuit breaker panel. The branch buses and associated wiring are protected by two extension bus circuit breakers located on the lower rear section of their opposite circuit breaker panel. This branch bus configuration is necessary to logically group circuit breakers together. As an example, the left and right engine instrument circuit breakers are grouped together on the same circuit breaker panel but are powered by different buses. All major airplane systems circuit breakers are located on the left (pilot) circuit breaker panel. Avionics, APU, and accessory circuit breakers are located on the right (copilot) circuit breaker panel.

If the battery switch is in BATT (battery relay closed) with the engines running and the generator switches in GEN, the starter-generators connect directly to the feed buses and supply

all DC power for the entire DC bus system. In this configuration, the generators keep the batteries charged (Figure 2-12).

PROTECTION

General

Two generator control units (GCUs) regulate, protect, and parallel the generators. Each unit controls a power relay connecting the generator to its DC feed bus (Figure 2-12), permitting the relay to close when the starter-generator is out of the start mode, the generator switch is in GEN, and the generator voltage equals or exceeds system voltage. A field relay, located in the GCU, allows or prevents field excitation within the generator. When open, this relay deprives the power relay solenoid of its ground, causing the power relay to also open. An internal feeder fault (short circuit) or an overvoltage, when sensed, causes the field relay to open. The relay also opens when the engine fire switch is activated. A reverse current (10% of total load) or undervoltage opens the power relay and thereby protects the 300-ampere current limiters and the generators.

The three parallel circuits between each DC feed bus and its extension bus are protected by three 80-ampere current limiters on the main bus and three 75-ampere circuit breakers on the corresponding cockpit circuit breaker panel. Two circuit breakers at each extension bus are circuit protection to the branch buses located behind the opposite circuit breaker panel. A 15-ampere circuit breaker on the right circuit breaker panel at the emergency branch bus is circuit protection to the emergency branch bus on the left circuit breaker panel. Various other circuit breakers at the junction box and the auxiliary junction boxes in the tail, protect against overloads.

Between each DC feed bus and the crossfeed bus, a 300-ampere fuse limiter protects the system from overloading. Loss of either current limiter causes the system to split and become two independent systems, a left and a right.

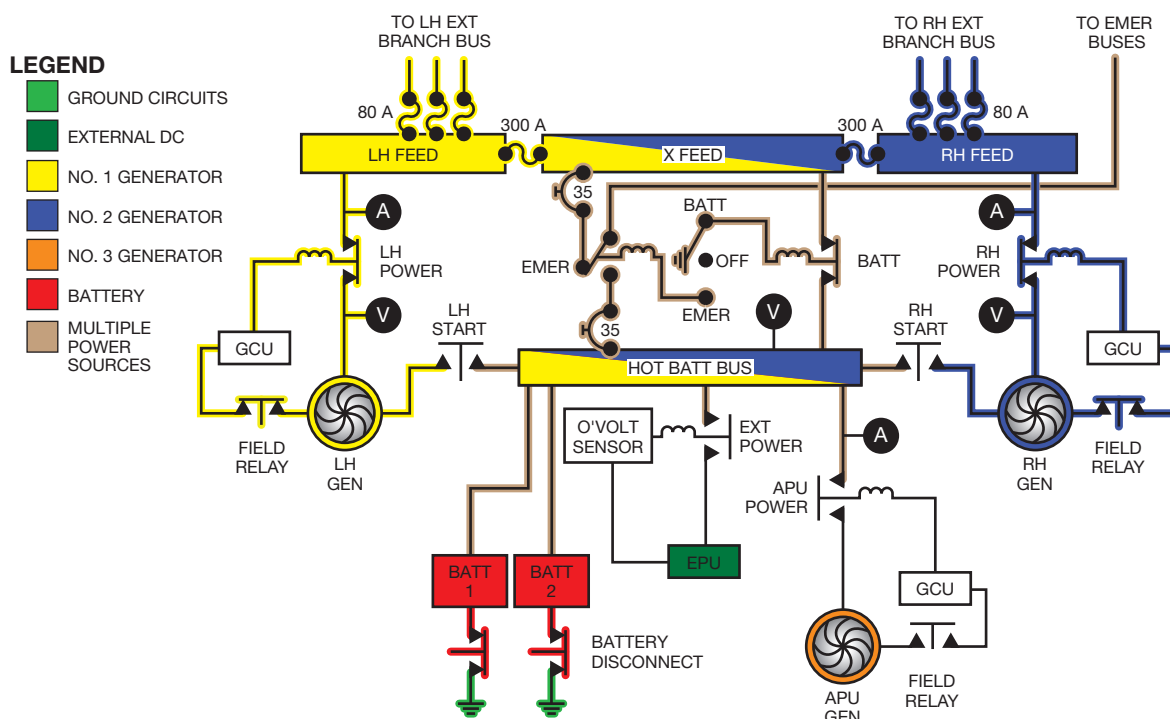
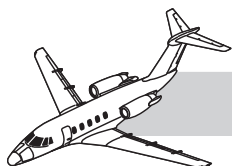


Figure 2-12. Electrical Circuit Condition—Battery Switch in BATT, Generators On Line

Generator-Assisted Start

When one generator is operating and connected to its main load bus, it is necessary to protect the 300-ampere current limiter on that side from the high amperage required to start the opposite engine. To provide this protection, the battery relay opens the circuit between the hot battery bus and the crossfeed bus during the second engine start. This prevents high amperage flow through that current limiter (Figure 2-13).

Overvoltage EPU

Should EPU voltage be excessive, an overvoltage sensor opens the external power relay, breaking the circuit to the hot battery bus.

External power disable relays also disconnect the EPU from the hot battery bus whenever a power relay closes, bringing a generator on the line.

CONTROL

Control of the DC power system is maintained through the battery switch and two generator switches. The battery switch has three positions: BATT, OFF, and EMER. With the switch in OFF, the hot battery bus is isolated from the other buses, and the emergency bus is connected to the crossfeed bus. The BATT position closes the battery relay, and the emergency bus remains connected to the crossfeed bus. In the EMER position, the battery relay is opened, isolating the hot battery bus from the rest of the system, but the emergency bus is connected to the hot battery bus.

The generator switches have three positions: GEN, OFF, and RESET. Placing the switch to GEN allows the GCU to close the power relay, connecting the generator to its DC feed bus. With the switch in the OFF position, the power relay does not close, and the generator does not assume any load. Placing the switch in the RESET position should close the generator field relay if it has opened.

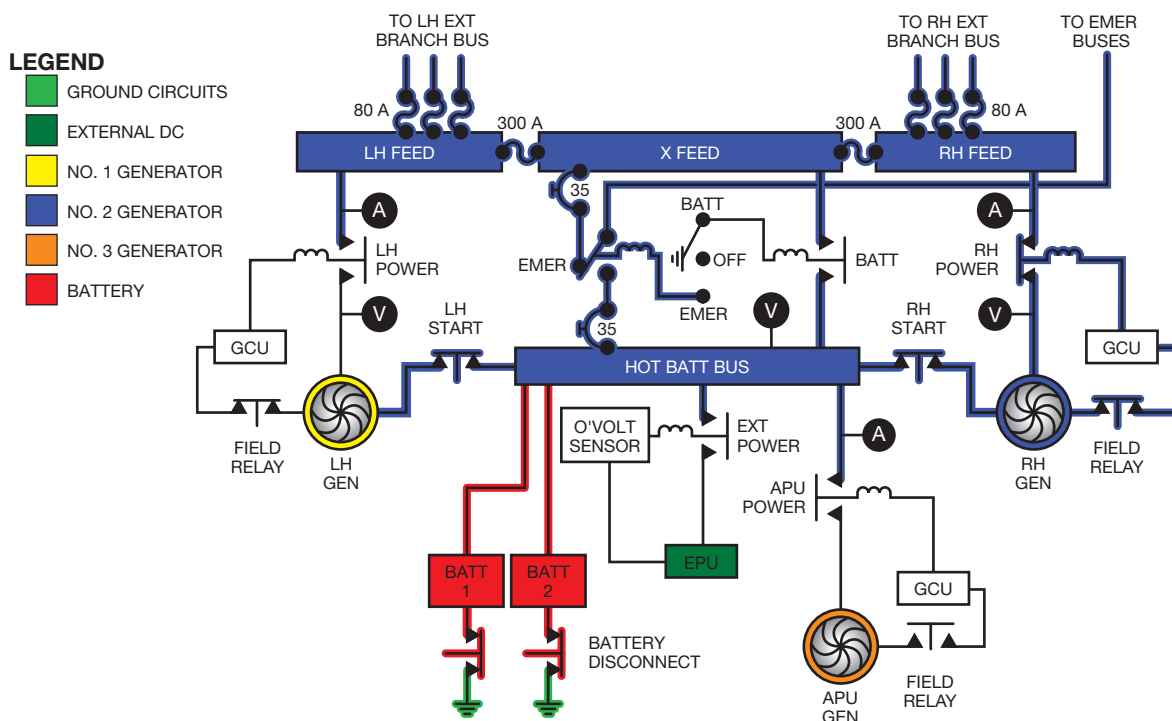
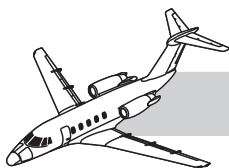


Figure 2-13. Electrical Circuit Condition—Generator-Assisted Start

On the center panel are two engine start buttons. When depressed, they apply DC power to the start circuit, which closes the start relay, allowing current to flow from the hot battery bus directly to the starter-generator. A starter disengage button, located between the starter buttons, can be used to open the start relay if manual termination of the start sequence is required.

Other switches such as the BATT DISC switch in the cockpit and the STARTER DISABLE switch in the baggage compartment are discussed in the Operations section of this chapter.

MONITORING

The DC electrical system is monitored by a voltmeter, two ammeters, GEN OFF LH–RH warning lights, BATT O’TEMP 1 or 2 warning lights, and an optional battery temperature readout (Figure 2-14).

When illuminated, a GEN OFF annunciator light indicates an open power relay, breaking the circuit between the generator and its feed bus, thus preventing the generator from

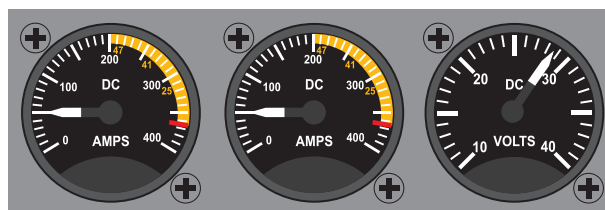


Figure 2-14. DC Electrical Indicators

accepting a load. If both GEN OFF annunciator lights are illuminated, the MASTER WARNING RESET switchlights on the instrument panel flash.

A voltmeter selector switch permits the pilot to read voltage from the hot battery bus and from a point between each generator and its power relay. It is spring-loaded to the hot battery bus. With the switch in BATT, the voltage on the hot battery bus can be read regardless of the battery switch position. The voltmeter indicates the highest voltage of the source providing power. When one generator is on line and the voltmeter selector switch is in either BATT or the corresponding generator



position, the voltmeter reads the generator's voltage. If the voltmeter selector switch is moved to read the opposite generator's output (the generator not connected to the buses), it indicates the voltage of that generator.

The ammeters read the current flow from their individual generators to the DC feed bus. During normal operation, their indication should be approximately equal ($\pm 10\%$ of total load). Amperage through the start relays does not indicate on the ammeter.

A temperature sensor in each battery initiates a steady BATT O'TEMP 1 or 2 light on the annunciator panel (with flashing MASTER WARNING RESET switchlights) when the battery temperature rises to 60°C . If the temperature continues to rise to 71°C , the BATT O'TEMP 1 or 2 light flashes (again, with flashing MASTER WARNING RESET switchlights).

An optional digital temperature gauge reads the temperature of each battery. At 60°C a yellow light on the gauge illuminates, and at 71°C a red light illuminates and the yellow light extinguishes. The lights are controlled by the gauge and are independent of the annunciator warning lights. When the rotary test knob is placed in the BATT TEMP position, the BATT O'TEMP 1/2 and the MASTER WARNING RESET switchlights flash, the yellow and red lights on the gauge illuminate, and the temperature displays read -188°C .

In each start button, a light illuminates when the start relay closes, completing a circuit between the hot battery bus and the starter-generator.

An open circuit breaker on the main or auxiliary junction boxes is indicated by illumination of the AFT J-BOX CB annunciator light. Illumination of the AFT J-BOX LMT light indicates failure of either the left or right 300-ampere current limiter.

OPERATION

NORMAL

Engine Starts

During the internal preflight cockpit inspection, the generator switches should be placed to GEN unless a start using an external power unit is intended. In this case the switches should be placed to OFF. The battery voltage should be checked for 24 volts minimum. During the first preflight inspection of the day, the batteries should be checked for signs of deterioration or corrosion and proper connection. Raising a door on the floor of the baggage compartment provides access to the batteries.

Battery Starts

Before starting the engines, the generator switches should be rechecked for proper position and battery voltage verified. The battery switch must be in the BATT position in order to provide power to the left and right extension buses to close the respective start relays.

Located on each extension bus is a 7.5-ampere circuit breaker labeled "LH" or "RH START." Power must be available through these circuit breakers in order to close the respective start relay and energize the start circuit. Either engine may be started first. When a start button is depressed, closure of the start relay is indicated by illumination of the light in the start button. At this time, the electric fuel boost pump is energized, the engine instrument and cockpit floodlights illuminate, and the ignition system is armed (ignition lights are not illuminated). Closure of the start relay connects the hot battery bus to the starter motor for engine rotation. At 10% turbine (N_2) rpm and indication of fan (N_1) rpm rotation, the throttle is brought from CUT OFF to the IDLE detent. Ignition is activated by a throttle switch, and a green light over the ITT tape illuminates, indicating electrical power to the exciter box. Fuel is introduced as the throttle is advanced, and combustion should occur within ten seconds and be evidenced by rising ITT. As the



engine accelerates, at 42 to 48% N₂ rpm, the start sequence is automatically terminated by a speed-sensing switch in the starter-generator. At termination, the electric boost pump and ignition deactivate, the start relay opens, the starter button light extinguishes, and the engine instrument and cockpit floodlights extinguish (Figure 2-15).

At start termination, the starter-generator reverts to generator operation, and the GCU closes the power relay to bring it on the line when generator voltage equals or exceeds system voltage. This is indicated when the associated GEN OFF annunciator light extinguishes, a load is shown on the associated DC ammeter, and the DC voltmeter reads 28.5 volts. At this point, the generator is supplying electrical power to the entire electrical system and recharging the battery.

Generator-Assisted Battery Start

For a subsequent engine start on the ground, the operating generator assists the battery in providing current to the starter motor. The operating engine must be stabilized above 61% N₂ rpm minimum to prevent the operating engine speed from decreasing below N₂ idle rpm during the second engine start. Amperage on the operating generator is allowed to drop below 200 amps. Upon activating the starter button, the electric boost pump operates, ignition is armed, engine floodlights illuminate, both start relays close, and the light in each start button illuminates. This indicates that both start relays are closed and there is a direct path of current flow from the operating generator through the hot battery bus to the starter as electrical assistance for the battery. The operating generator must be on line (power relay closed) to provide generator assist during the second engine start.

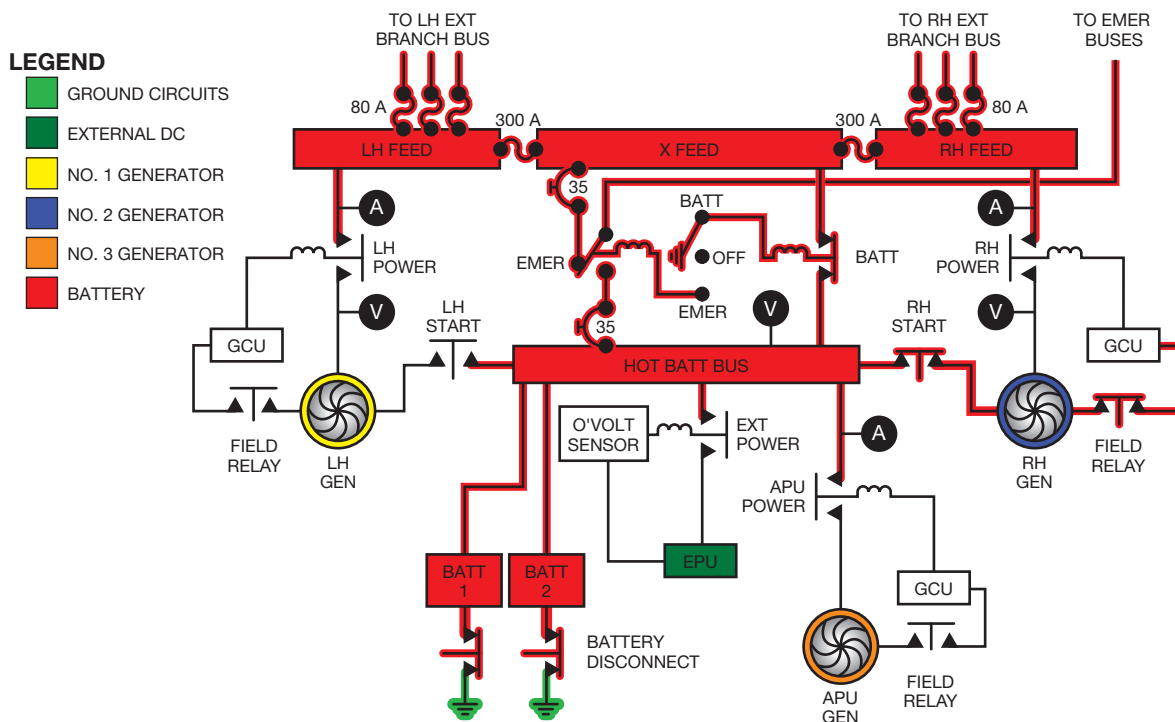


Figure 2-15. Electrical Circuit Condition—Right Engine, Battery Start



To prevent high amperage flow through the assisting generator's 300-ampere current limiter, the battery relay automatically opens. This prevents the high current from seeking an alternate path to ground and disabling the current limiter. Each starter normally requires an excess of 1,000 amps to initiate engine rotation. The generator power relay remains open on the starting engine side and prevents high amperage from flowing through that 300-ampere current limiter. After the second engine start sequence is terminated (by action of its respective starter-generator speed-sensing switch) and the generator comes on line, both GEN OFF annunciator lights are extinguished and the DC ammeters indicate equal load.

In-Flight Engine Starts

Generator-assist capability is disabled in flight. This is to prevent power drain from the main DC buses and to protect the operating generator during an engine start utilizing the start button. Either main gear squat switch prevents the opposite start relay from closing during starter engagement in flight.

Depressing the start button to restart an engine in flight is strictly a battery start, and battery limitations must be observed. Only one start button light illuminates. The battery relay still opens to protect the 300-ampere current limiter on the side of the operating generator.

EPU Starts

An external power unit (EPU) may also be used for engine starts. Prior to use, the unit should be checked for a voltage of 28.5 volts and the capability of delivering at least 1,000 amps (but not more than 2,000 amps).

The battery disconnect relays open when the EPU is plugged in, thus preventing battery discharge during engine starts (Figure 2-16). When external starts are intended, check the voltmeter for 28.5 volts, and be sure that the generator switches remain in the OFF position until both engines have been started. All monitoring procedures remain the same as battery starts except that the generators do not come on the line after each start.

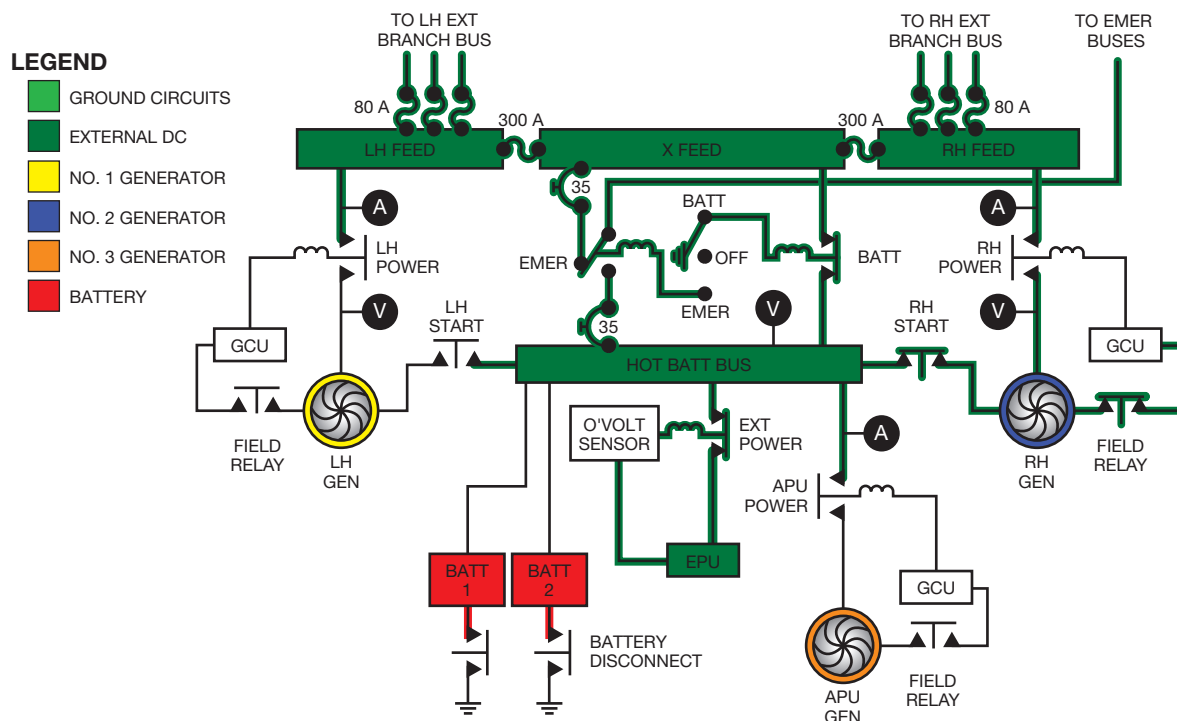


Figure 2-16. Electrical Circuit Condition—Right Engine, EPU Start



Provided the generator switches are in OFF, the second engine start may be initiated immediately after the first start sequence is terminated. Only the respective start button illuminates during EPU starts. After both engines are started, place the generator switches to GEN and disconnect the EPU.

Preflight Electrical System Check

After both engines are started and the generators are on line, the electrical system should be checked for proper operation prior to flight as follows:

Check that both generators are accepting their share of the total electrical load equally by observing both DC ammeters. They should display equal amps within 10% of total load.

The generators may be checked individually by turning off each generator switch one at a time. When the switch is off, observe the on-line generator accepting the total amperage load and read the voltage. Check the off-line generator voltage by placing the voltmeter selector switch to that generator position. It should indicate 28.5 volts. Proper voltage indications, load acceptance, and sharing during these checks should validate proper operation of each generator and associated GCU.

All other cockpit and annunciator lights that pertain to the electrical system should be extinguished.

ABNORMAL

Monitoring the ammeters may provide the pilot indication of impending generator problems. A difference in amperage readings indicates that the generators are not parallel. When a GEN OFF light illuminates on the annunciator panel, a check of the voltmeter indicates whether the field relay or only the power relay has opened. An open field relay could be caused by a feeder fault (short circuit), overvoltage, or actuation of the engine fire switch. A tripped field relay is indicated by near-zero generator voltage and can possibly be reset with the gen-

erator switch. An undervoltage or reverse current causes the generator control unit to open the power relay. If any voltage is observed on the voltmeter when that generator is selected with the voltage selector switch, generator reset is not probable.

In-flight operations with one generator should not pose a serious problem as long as single-generator amperage load restrictions are observed based on airplane altitude.

Loss of Both Generators

In-flight operations with the battery supplying all the electrical power requirements pose a serious problem due to rapid battery discharge. Placing the battery switch to EMER sheds the load requirements of the main DC buses (Figure 2-17). This is done by opening the battery relay, thus isolating the hot battery bus from the crossfeed bus. The emergency bus and emergency branch bus are still connected to the hot battery bus through the emergency relay. In this configuration, the battery provides electrical power for minimum flight-sustaining night IFR equipment, and it should last approximately 30 minutes.

In addition to essential DC electrical power supplied by the hot battery bus and the emergency DC buses, essential 26-volt AC power is supplied by the pilot C14D gyro inverter for the following items (SNs 0001—0178):

- ADC/pilot altimeter
- HSI 1 (mechanical only)
- No. 1 NAV (EFIS only)
- No. 1 Mach/airspeed

The pilot C14D gyro is powered from the emergency DC bus.

Battery Overheat

Battery overheat could result from an excessive amount and rate of charge or internal battery damage. The greatest damage which can result from a battery overheat lies in the possibility of runaway heating, in which internal failures cause the heat to continue building out of control.

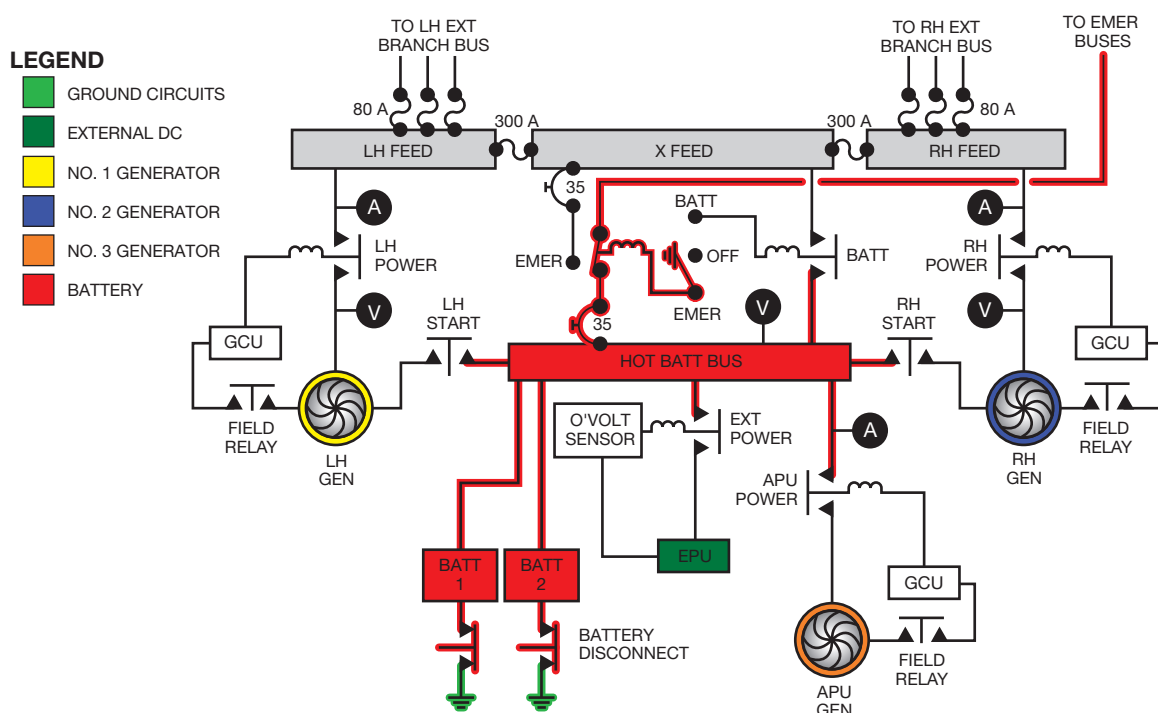
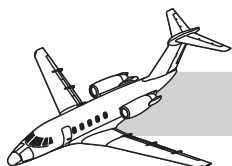


Figure 2-17. Electrical Circuit Condition—Generators Off Line, Battery Switch in EMER

Battery overheat is indicated initially by a steady BATT O’TEMP light on the annunciator panel and flashing MASTER WARNING RESET switchlights which can be cancelled. Battery temperature has reached 60°C. The battery temperature gauge, if installed, can verify the temperature and, above 60°C, displays a yellow light. Continued rising temperature (71°C) causes the BATT O’TEMP light to flash and reilluminates the MASTER WARNING RESET switchlights. When the gauge reaches 71°C, the yellow light extinguishes and a red light illuminates. Whenever an overheat condition exists, select the overheated battery with the battery disconnect switch to remove the battery from charging power (Figure 2-18).

On single-battery aircraft, place the battery switch in OFF to isolate the battery. (Refer to the checklist/*AFM* for the proper procedures.) Aircraft configured with a single battery do not have a battery disconnect switch.

Bus Disabling Procedures

Should it be necessary to disable the circuit breaker panel at the pilot position, it can be accomplished by pulling the three 75-ampere circuit breakers labeled “LH FEED BUS” and, on the copilot circuit breaker panel, the two circuit breakers labeled “LH CB PNL PWR” and the 15-ampere EMER CROSSOVER circuit breaker. The first three circuit breakers disconnect the left main bus extension, the next disconnects the branch buses from the right extension bus, and the last disconnects the emergency branch bus. A similar procedure can be used to disable the copilot panel. The three 75-ampere circuit breakers on the panel and the two branch circuit breakers on the pilot panel should be pulled. In addition, the following emergency item breakers on the copilot panel should be pulled (SNs 0001—0178): EMER CROSSOVER, STDBY ATT IND, DIRECT GYRO 1, COMM 1, NAV 1, ADC, H TRIM ADVISE: RMI 1 (EFIS), NAV XFER. (SNs 0179 and subsequent): DADC1, COMM1, NAV1, RMI1, AUDIO 1/2, PILOT

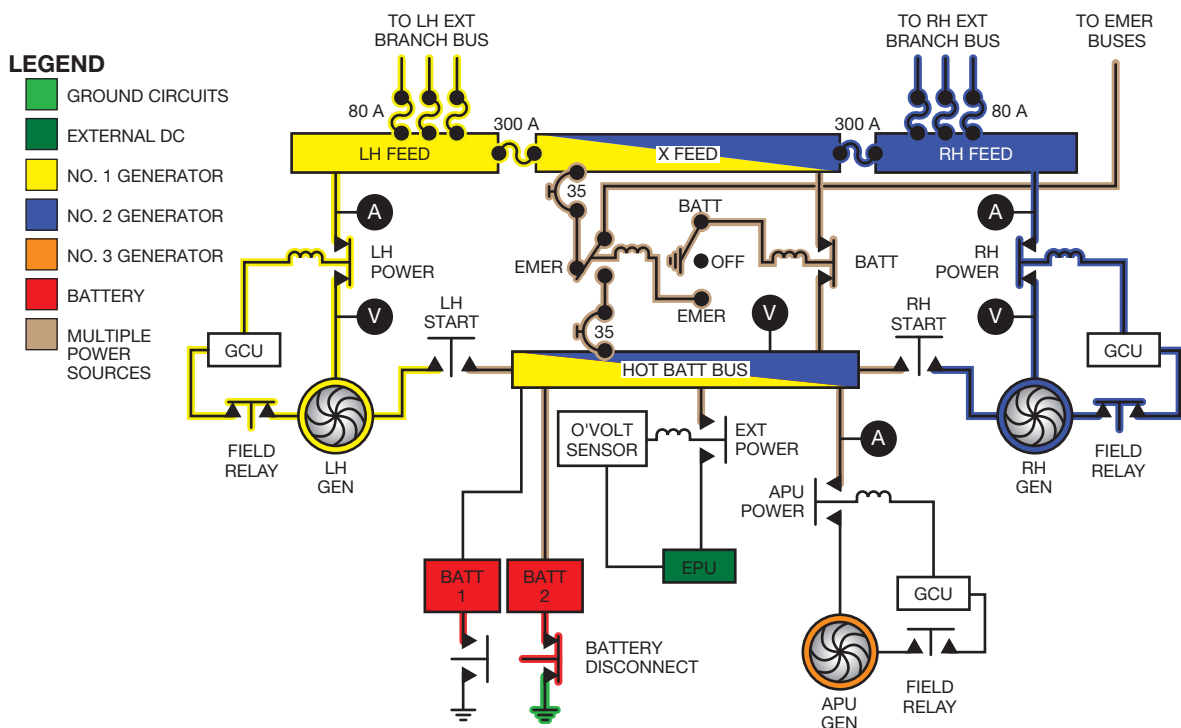
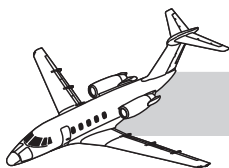


Figure 2-18. Electrical Circuit Condition—No. 1 Battery Disconnected

ALT, H TRIM ADVISE, AHRS2, STDBY NAV/COM—PRIMUS II only, RMU—PRIMUS II only. In addition, the AC power switch should be positioned to OFF (SNs 0001—0178).

NOTE

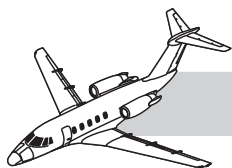
When the three 75-ampere main bus circuit breakers are pulled, the branch buses to the opposite circuit breaker panel are also disabled.

NOTE

Pulling the EMER CROSSOVER circuit breaker disables the emergency branch bus on the pilot circuit breaker panel.

300-Ampere Feed Bus Current Limiter Failure

Failure of a 300-ampere current limiter can be recognized by illumination of the AFT J-BOX LMT light on the annunciator panel. If the generator switch is turned off for the side of the blown current limiter, a complete loss of electrical power results on all that side's main DC buses (feed bus, extension bus, and branch bus). Loss of power to one extension bus can be easily recognized by the steady illumination of a MASTER WARNING RESET switchlight for the same side of the cockpit as the failed extension bus. The light cannot be reset in these circumstances. Off flags come into view on the engine instruments for the same side except N_1 and ITT to further verify a loss of power. Failure of a 300-ampere current limiter prior to starting engines prevents starting an engine on the side of the failure. DC power through the LH or RH START circuit breaker on the respective extension bus is required to start the respective engine.



Electrical System Engine Start Malfunctions

If the speed-sensing switch fails to terminate a start sequence, the starter disengage button can be used to terminate the start. This action prevents any system component damage. The GCU does not permit the generator to come on line until the start sequence has terminated.

In the event that the start relay does not open at start termination, the light in the start button remains illuminated. The starter disengage button can be used in an attempt to open the relay, but probably not effectively. With the engine running, the STARTER DISABLE switch inside the upper forward corner of the baggage compartment door should be activated. This electrically disconnects the batteries, and the engine may be shut down without being motored by the starter. The battery quick-disconnects should now be opened and the STARTER DISABLE switch returned to the normal position (guard closed). Activation of the switch causes battery power to open the disconnect relays in the battery ground leads and gradually discharges the batteries.

NOTE

If the opposite engine is operating with its generator on line, you can protect the current limiter on the opposite side by placing the battery switch to OFF prior to shutting down the affected engine.

**AC POWER: SNs
0001-0178, 0200-0202,
0207-0241**

GENERAL

Normally, AC power is provided by two 350 volt-ampere static inverters. The inverters received input power from the main DC system and produce both 115-volt and 26-volt AC

power. Each inverter powers a 115-volt and 26-volt bus so that, in normal operation, there are two independent systems (Figure 2-19). In the event of inverter failure, the operating inverter automatically powers all AC buses.

PROTECTION

The DC power source to the inverters is protected by circuit breakers on the left and right circuit breaker panels. Internal current and voltage-sensing circuits provide protection for the inverter and shut off an inverter if it performs outside of its design parameters. Individual buses are protected by four circuit breakers on the right circuit breaker subpanel (Figure 2-20).

CONTROL

The inverter switch (Figure 2-21) is mounted to the right of the master avionics switch under the legend "AVIONICS POWER." The two switch positions are INV ON and OFF. The switch controls both inverters and activates them when positioned to INV ON.

MONITOR AND TEST

The systems are monitored by INVERTER FAIL 1 or 2 annunciator lights. In the event inverter power is lost, indicating failure or automatic shutdown due to improper voltage, the associated light illuminates. After the inverters are turned on, they may be tested with a test switch located on the right of the inverter switch. The test switch has three positions: INV 1 TEST, OFF, and INV 2 TEST. With the switch held in the INV 1 TEST position, the No. 1 inverter is turned off and the INVERTER FAIL 1 light should illuminate. The test is repeated for inverter No. 2. During the test, all AC equipment continues to receive power.



CITATION 650 SERIES PILOT TRAINING MANUAL

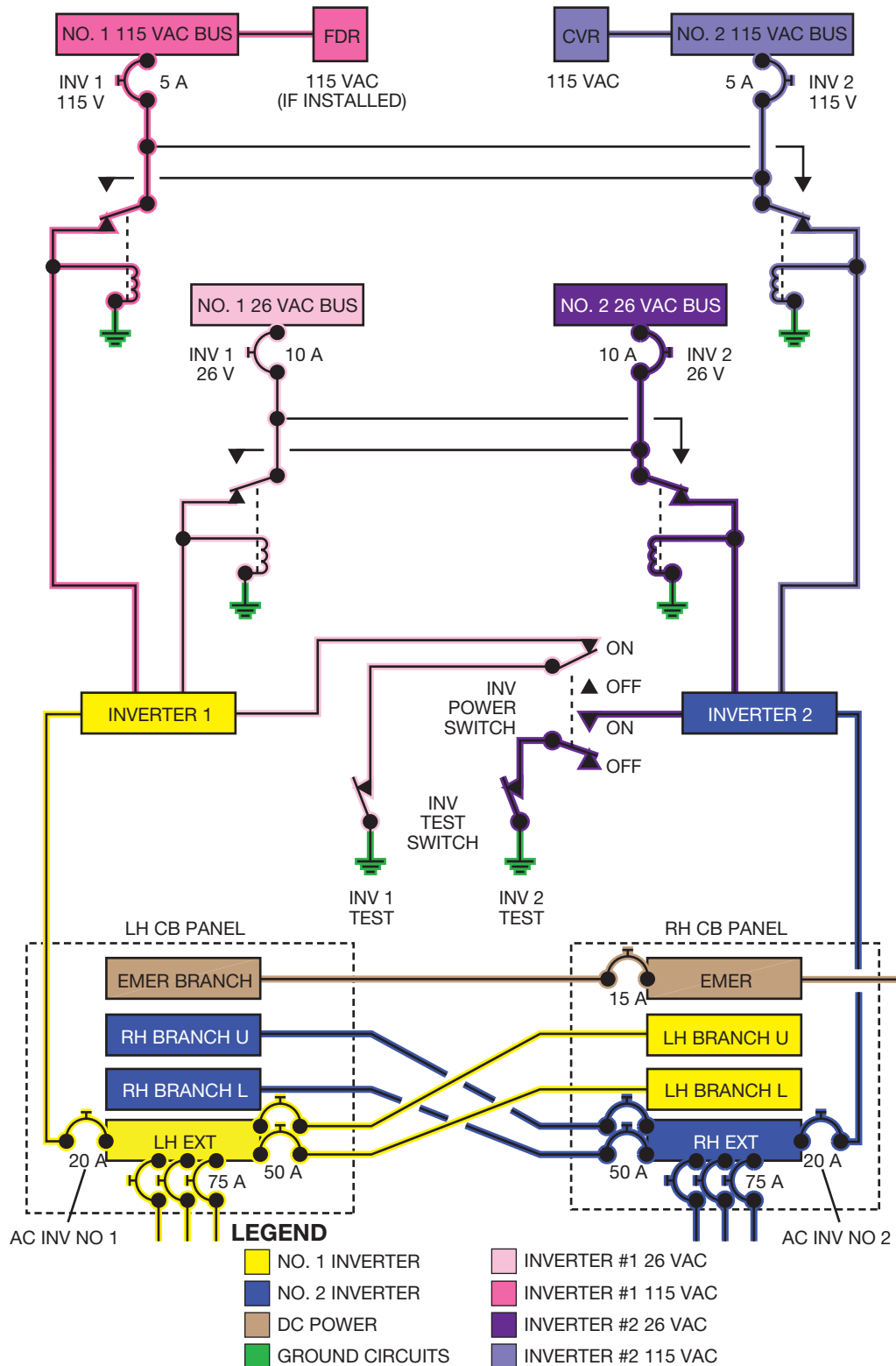


Figure 2-19. AC Power Distribution: SNs 0001—0178, 0200—0202, and 0207—0241

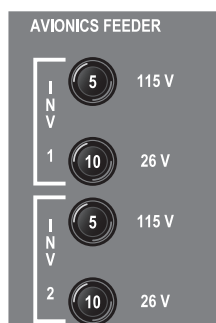
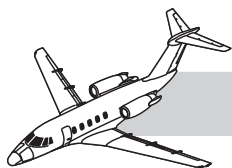


Figure 2-20. AC Circuit Breaker Subpanel: SNs 0001—0178, 0200—0202, and 0207—0241

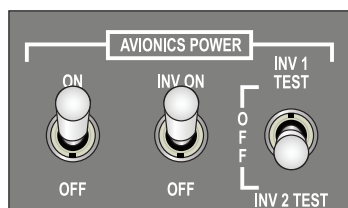


Figure 2-21. AC Controls: SNs 0001—0178, 0200—0202, and 0207—0241

NORMAL OPERATION

In normal operation, the AC power switch is turned on and the system tested. Each inverter powers its own set of buses. In the event of failure, the buses powered by the failed inverter automatically transfer to the operating inverter.

NOTE

There are two AC ammeters and an AC voltmeter on the right instrument panel. These instruments are not, and cannot be, connected to and used with this system. They are used for monitoring the horizontal stabilizer leading-edge anti-ice systems. Their use is discussed in Chapter 10 of this manual.

ABNORMAL OPERATION

In the event of a single inverter failure, the indication is illumination of the associated

INVERTER FAIL annunciator. In the event that both inverters fail, the MASTER WARNING RESET switchlights illuminate as well as both INVERTER FAIL annunciators. In either case, refer to the appropriate single or dual Inverter Fail checklist in the Emergency/Abnormal Checklist. In the case of a single inverter failure the remaining inverter will power all AC systems. In the case of a dual inverter failure the emergency AC items can receive backup AC power from the C-14D directional gyro inverter (Figure 2-22 and 2-23). In the event of a failure of the AC system, check the circuit breakers and comply with the appropriate checklist in the Emergency/Abnormal Checklist.

In the event of a dual generator failure, following the appropriate procedure in the Emergency/Abnormal Checklist will direct selection of the battery switch to EMER in order to shed nonessential buses and power the emergency items from the hot battery bus and emergency buses. Placing the batter switch in EMER also enables the emergency AC items to receive AC power from the C-14D directional gyro inverter.

AC POWER: SNs 0179–0199, 0203–0206, 7001–7119

GENERAL

AC power is provided by two 250 VA static inverters. The inverters receive input power from the main DC system and produce both 115 and 26 VAC power. Each inverter powers a 26 VAC bus so that in normal operation there are two independent systems (Figure 2-24). In the event of an inverter failure, the operating inverter can power both 26 VAC buses. 115 VAC power from the No. 1 inverter is supplied directly to the optional flight data recorder.

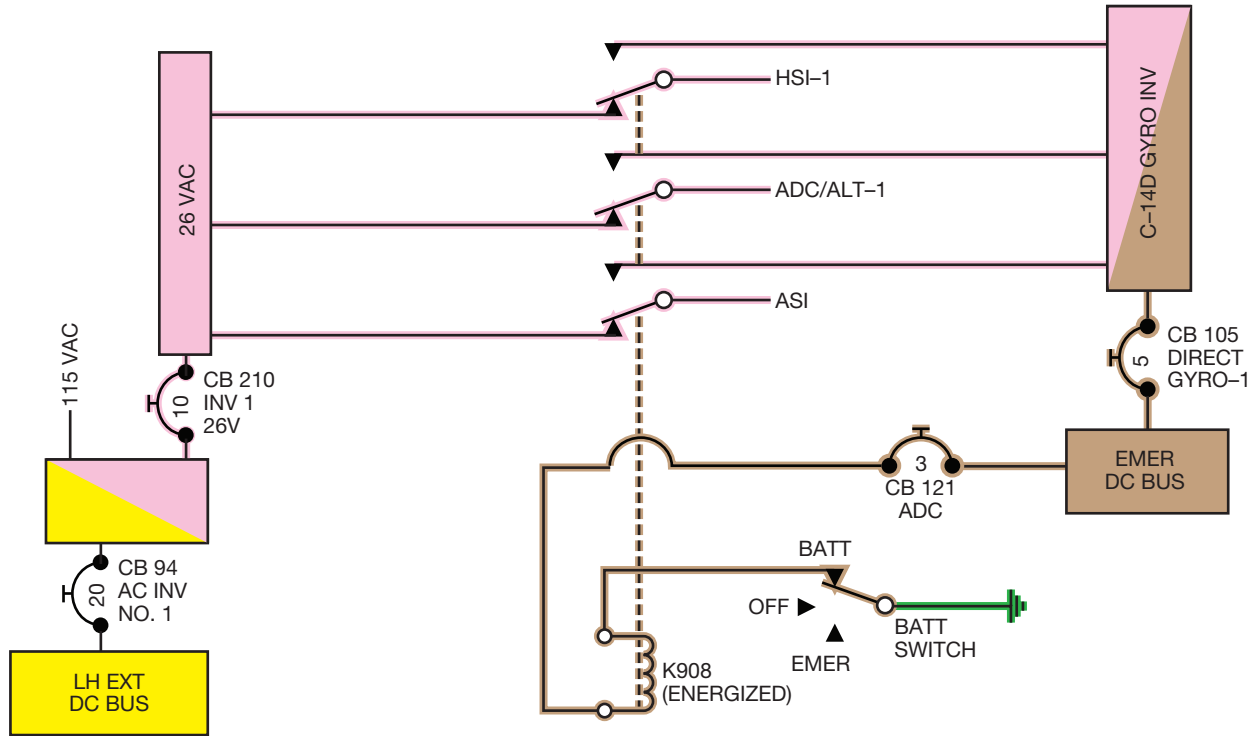


Figure 2-22. Emergency AC Power: SNs 0001 – 0066 (1 of 2)

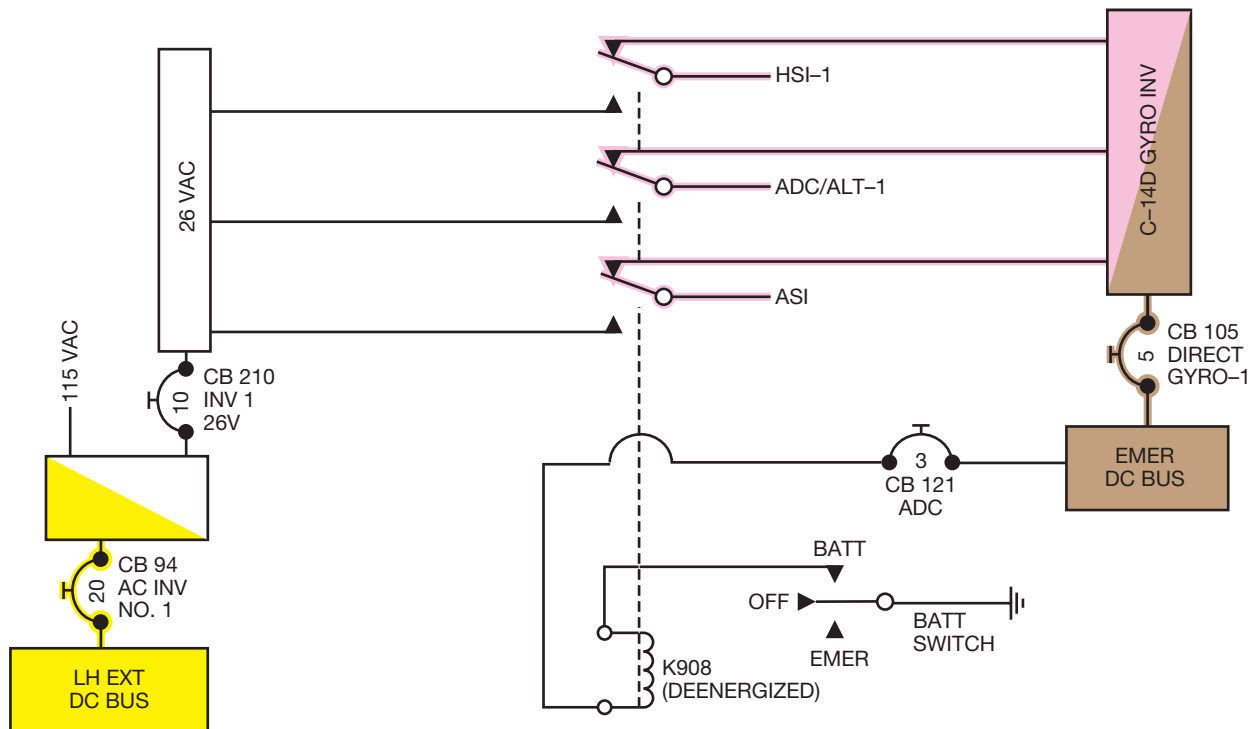


Figure 2-22. Emergency AC Power: SNs 0001 – 0066 (2 of 2)



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**2 ELECTRICAL POWER
SYSTEMS**

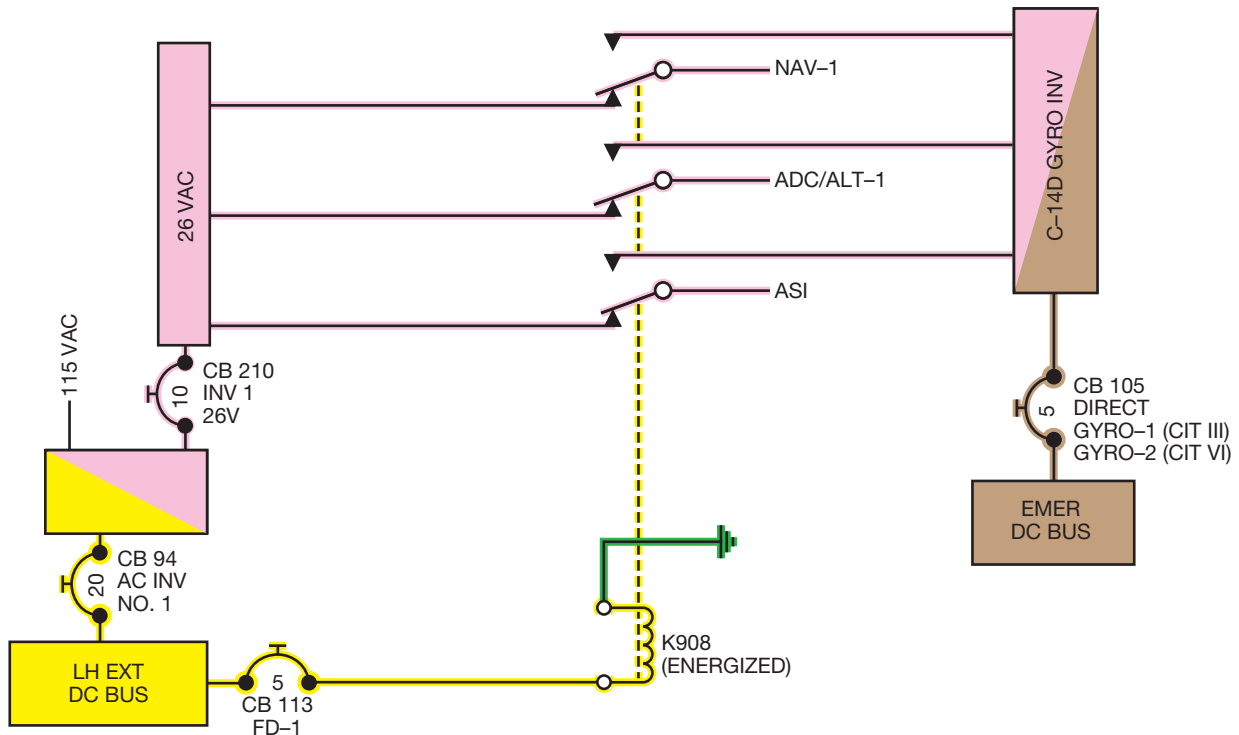


Figure 2-23. Emergency AC Power: SNs 0001 – 0066 (W/EFIS), 0067 – 0178, 0200 – 0202, and 0207 – 0241 (1 of 2)

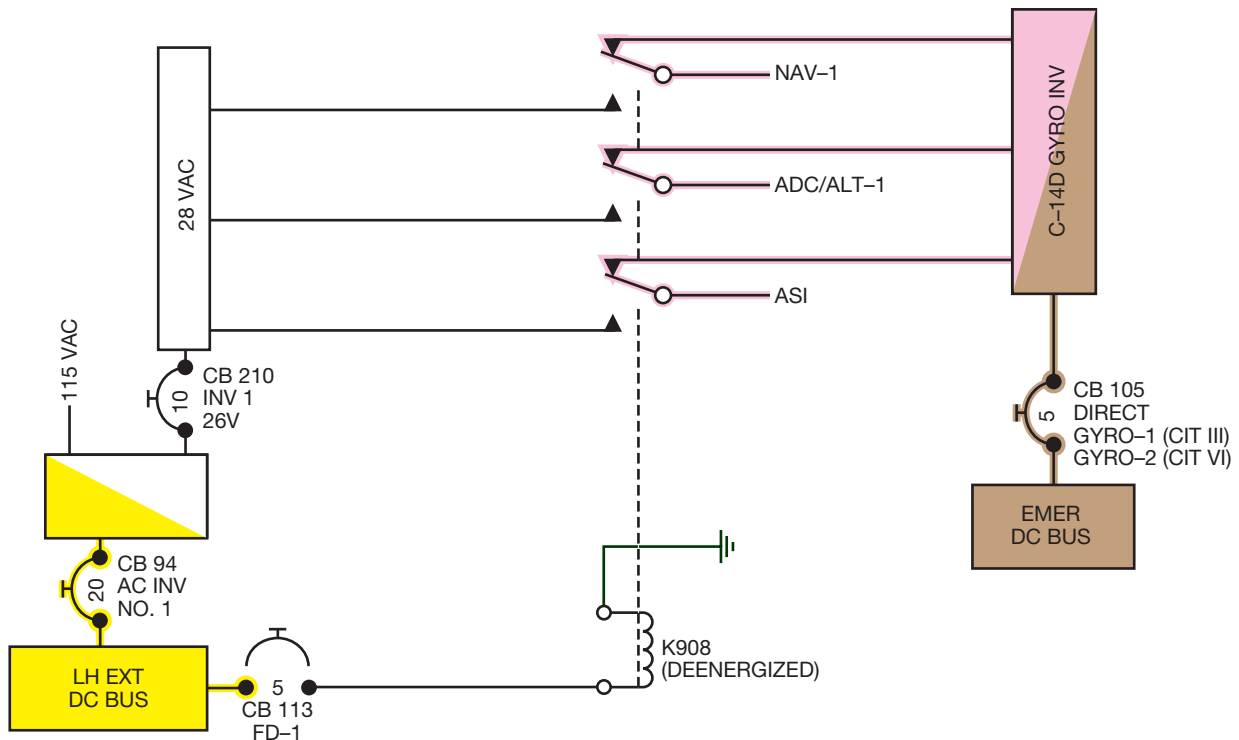


Figure 2-23. Emergency AC Power: SNs 0001 – 0066 (W/EFIS), 0067 – 0178, 0200 – 0202, and 0207 – 0241 (2 of 2)

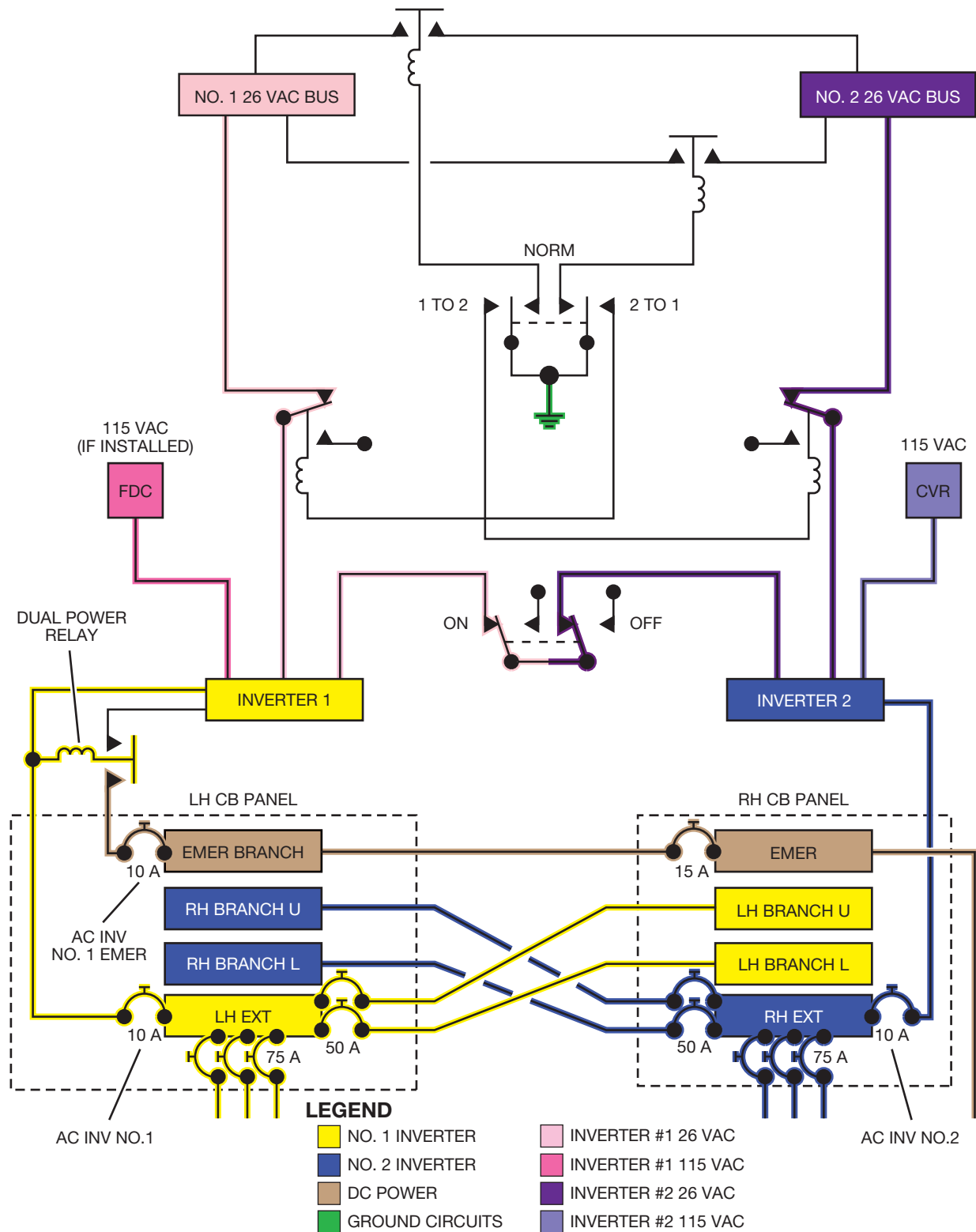
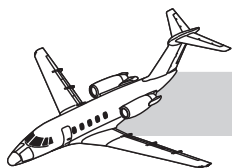


Figure 2-24. AC Power Distribution: SNs 0179–0199, 0203–0206, and 7001–7119



PROTECTION

The DC power source to the inverters is protected by 10-ampere circuit breakers on the left and right circuit breaker panels. The left circuit breaker panel has two 10-ampere circuit breakers for the No. 1 inverter: “AC INV NO.1” connected to the left extension bus and “AC INV NO. 1 EMERG” connected to the emergency branch bus. If left main DC bus power is lost or the AC INV NO 1 circuit breaker opens, then a relay deenergizes closed to allow the emergency branch bus to provide 28 VDC power to the No. 1 inverter via the AC INV NO. 1 EMERG circuit breaker.

CONTROL

Two AC control switches (Figure 2-25) are located on the center tilt panel in the AVIONICS POWER section. The left ON-OFF switch controls both AC and DC power for all avionics. The right AC XOVER switch is used during inverter failure. Placing the XOVER switch to the 2 TO 1 position disconnects the No. 1 inverter from the No.1 26 VAC bus and connects this bus to the No. 2 26 VAC bus, allowing the No. 2 inverter to power both buses. Placing the XOVER switch to the 1 TO 2 position reverses the order, allowing the No. 1 inverter to power both 26 VAC buses and drop the No. 2 inverter off line (Figure 2-24). Each inverter supplies power to its own respective AC bus when the AC XOVER switch is in the NORM position.

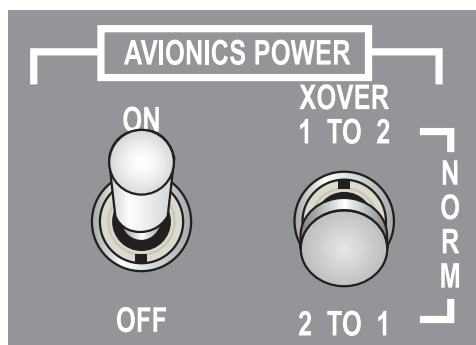


Figure 2-25. AC Power Switches: SNs 0179–0199, 0203–0206, and 7001–7119

MONITOR AND TEST

The systems are monitored by INVERTER FAIL 1 and 2 annunciator lights. In the event inverter power is lost, indicating failure or automatic shutdown due to improper voltage, the associated light illuminates. If this malfunction should occur, place the XOVER switch to the proper position to allow the operating inverter to power both 26-VAC buses. After the master avionics switch is turned to ON, the AC system may be tested by moving the XOVER switch from NORM to the 1 TO 2 position and confirming that the INVERTER FAIL 2 light illuminates and no flags are in view on the pilot and copilot instrument panels. Repeat this test for the No. 1 inverter by switching to the 2 TO 1 position (Figure 2-25).

NORMAL OPERATION

In normal operation, the AC power switch is turned ON, and the system is tested. With the XOVER switch in NORM, each inverter powers its own 26-VAC bus and provides direct 115 VAC to the flight data recorder, if installed. The INVERTER FAIL 1 and 2 annunciator lights should remain extinguished unless the system is tested or an inverter malfunction exists.

ABNORMAL OPERATION

If an inverter should fail, the indication is illumination of the associated INVERTER FAIL 1 or 2 light. If this malfunction should occur, place the XOVER switch to the operating inverter; i.e., for a No. 1 inverter failure, switch to 2 TO 1; for a No. 2 inverter failure, switch to 1 TO 2. This allows the operating inverter to power both the No. 1 and No. 2 26-VAC buses.

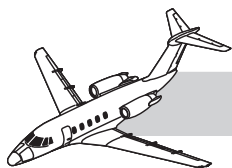


If both main DC generators fail and the battery switch is placed to EMER (loss of main DC power), the No. 2 inverter is lost, but the No. 1 inverter automatically switches to the emergency branch bus to power the pilot primary flight instruments, which include the following:

- Pilot OBI
- Pilot RMI/PN 101
- Pilot Mach/airspeed indicator
- Pilot DADC altimeter
- Pilot analog IVSI (optional)

In the event that both inverters fail, the MASTER WARNING RESET switchlights illuminate as well as both INVERTER FAIL lights. In this case, all AC-powered equipment is inoperative. The following flight instruments remain operational:

- Pilot EADI
- Pilot EHSI
- Pilot OBI
- Standby attitude indicator
- Pilot altimeter (pneumatic only)
- Pilot standby airspeed indicator
- Autopilot (only holds aircraft altitude at engagement)
- Copilot EADI
- Copilot EHSI
- Both RMI needles (ADF only relative bearing)

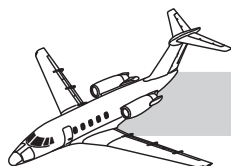


QUESTIONS

1. With the batteries as the only source of power and the battery switch in the OFF position, power is on the:
 - A. Hot battery bus and crossfeed bus.
 - B. Emergency bus and hot battery bus.
 - C. Hot battery bus.
 - D. Emergency bus, crossfeed bus, and hot battery bus.
2. With the batteries as the only source of power and the battery switch in the EMER position, power is on the:
 - A. Hot battery bus and crossfeed bus.
 - B. Emergency bus and hot battery bus.
 - C. Emergency bus only.
 - D. Emergency bus, crossfeed bus, and hot battery bus.
3. With the battery as the only source of power and the battery switch in the BATT position:
 - A. Only the hot battery, crossfeed, and feed buses receive power.
 - B. All buses are powered except the emergency bus.
 - C. Only the left and right DC feed buses are powered.
 - D. All DC buses are powered.
4. With only the hot battery and emergency buses powered, the inoperative equipment is:
 - A. NAV 2.
 - B. Right ITT and right N₁.
 - C. Left ITT and left N₁.
 - D. Both A and B.
5. The correct statement is:
 - A. With external power connected and the battery switch in OFF, all DC buses are powered from the EPU.
 - B. The DC voltmeter reads battery voltage with the battery switch in OFF.
 - C. With external power connected and the battery switch in OFF, all DC buses are powered by the EPU except for the battery itself.
 - D. The battery charges with the generators on the line, regardless of battery switch position.
6. A battery start on the ground:
 - A. Is normally terminated by the pilot with the starter disengage button
 - B. Requires that the boost pumps and ignition switches be turned on before the start button is depressed
 - C. Requires a minimum of 50% N₂ on the operating engine prior to starting the second engine
 - D. Is normally terminated by speed sensing on the starter-generator
7. The incorrect statement is:
 - A. The light inside the starter button indicates that the start relay has closed.
 - B. The generator switches are placed in the OFF position for an EPU start.
 - C. The battery switch is placed in the OFF position for an EPU start.
 - D. A failed left 300-ampere current limiter prevents starting of the left engine.



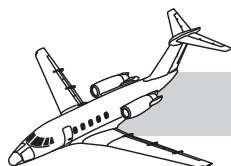
8. For a generator-assisted start:
- A. It is necessary to observe a drop in amperage to below 150 amps before depressing the second start button (to protect the 300-ampere current limiter).
 - B. The battery switch must be placed in OFF to protect the current limiter.
 - C. The start is terminated by the electronic fuel computer.
 - D. The operating engine is set to above 61% N₂ rpm.
9. Placing the battery switch in EMER with the generators on the line
- A. Does not cause the loss of any buses.
 - B. Causes the emergency bus to transfer to the crossfeed bus.
 - C. Still provides charging power to the battery.
 - D. Should result in a battery voltage of 28.5 VDC.
10. The correct statement regarding the GEN OFF LH–RH annunciator light is:
- A. Illumination of one light triggers the MASTER WARNING RESET switchlights.
 - B. It illuminates whenever the power relay is open.
 - C. It indicates that both the power and field relays have opened.
 - D. It indicates that only the field relay has opened.
11. Ignition during the start sequence
- A. Occurs immediately when the start button is depressed.
 - B. Is initiated automatically by the speed-sensing switch at 10% N₂ rpm.
 - C. Occurs when the throttle is brought to IDLE.
 - D. Is terminated by the fuel computer.
12. When the BATT O’TEMP 1 or 2 light comes on steady:
- A. Illumination of one light triggers the MASTER WARNING RESET switchlights.
 - B. If the light begins to flash, the battery is cooling down.
 - C. Place the battery switch to EMER.
 - D. The MASTER WARNING RESET switchlights do not illuminate until the BATT O’TEMP light begins to flash.
13. If the battery switch is in OFF in flight with both generators on the line:
- A. The emergency bus is powered by the hot battery bus.
 - B. The voltmeter does not read hot battery bus voltage.
 - C. All buses lose power except the hot battery and emergency buses.
 - D. The hot battery bus voltage drops to about 24 volts.
14. Illumination of the INVERTER FAIL 1 or 2 annunciator light:
- A. Triggers the MASTER WARNING RESET switchlights.
 - B. Indicates inverter voltage fluctuations.
 - C. Indicates only that the respective inverter is not operating.
 - D. Only indicates a loss of DC power to the inverter.
15. Illumination of both INVERTER FAIL 1 and 2 annunciator lights:
- A. Triggers the MASTER WARNING RESET switchlights.
 - B. Indicates that all flight instruments are lost.
 - C. Indicates that the weather radar functions are without AC power.
 - D. Indicates that inverter voltage can be read on the AC voltmeter.



CHAPTER 3 LIGHTING

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CHAPTER 3 LIGHTING



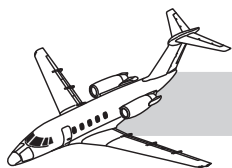
INTRODUCTION

Lighting on the Citation 650 series aircraft illuminates the cockpit area and all flight instruments, most of which are internally lighted. Floodlights provide general illumination of the cockpit. Map lights are at the pilot and copilot positions. The instrument panel is made of a luminescent material for easier viewing of instrument legends. Passenger advisory lights are in the cabin area and emergency lights are available to illuminate the exits if required. Various standard and optional lights illuminate the exterior.

GENERAL

Aircraft lighting includes interior lighting, exterior lighting, tailcone maintenance compartment lighting, and aft baggage compartment lighting. Interior lighting consists of cockpit, cabin, and emergency lighting. Cockpit lighting consists of instrument panel lights, electroluminescent panels, floodlights, and map lights. An optional indirect light strip

on the bottom face of the glareshield is available. Cabin lighting consists of indirect fluorescent lights, passenger reading lights, an aft compartment light, and advisory signs. Exterior lighting consists of lights for navigation, anticollision, wing inspection, taxiing, recognition and ground recognition, as well as tail floodlights and landing lights.



Tailcone maintenance compartment lights are for maintenance and preflight purposes. Aft baggage compartment lights are available for baggage loading and unloading areas and for preflight purposes.

INTERIOR LIGHTING

Interior lighting is in the cockpit, cabin, aft baggage, and tailcone compartments. Electroluminescent panels, floodlights, postlights, and internal lighting illuminate all cockpit instruments on the cockpit instrument panel, side consoles, and center pedestal. Electroluminescent panels illuminate function positions of switches and controls.

Secondary lighting includes rheostat-controlled floodlights, overhead map lights, and ice detection lights. Cockpit lighting controls are shown in Figure 3-1.

COCKPIT LIGHTING

Cockpit Floodlights

Two cockpit floodlights overhead near the center of the flight compartment (Figure 3-2) provide cockpit illumination and emergency lighting for the instrument panel. A rheostat on the left side PANEL LIGHT CONTROL

subpanel (Figure 3-1) labeled FLOOD LTS or FLD LTS controls floodlight intensity. During engine start, the floodlights illuminate at full intensity when the engine start circuit is energized. During engine start only, the floodlights are powered by the emergency light battery packs and their intensity cannot be adjusted with the rheostat.

Two small lights on the underside of the fire tray below the center glareshield illuminate the engine instruments during engine starts. This compensates for dimming of the internally lighted instruments when the start cycle places a heavy draw on the electrical system. The lights are powered from the emergency battery pack during the start. Other than during engine start, intensity of the cockpit floodlights is controlled with the FLOOD LTS or FLD LTS rheostat. These lights are normally powered by the emergency DC bus.

Control Panel Lights

Control panel lighting is provided by electroluminescent (EL) light panels, which feature white lettering on a gray background for easier viewing (Figure 3-1). The EL rheostat controls illumination intensity. The panels are activated by the DAY/NITE DIM ON switch. ELs are used on:

- CB panels
- Switch panel

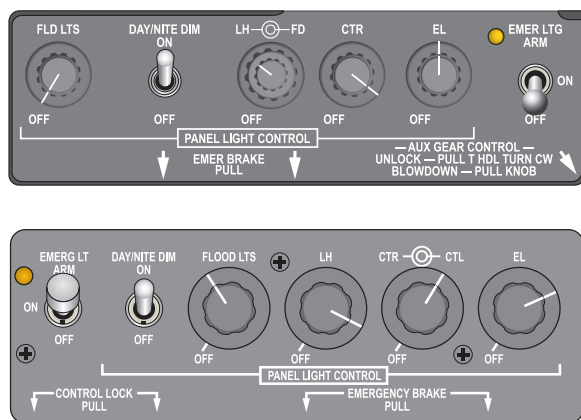
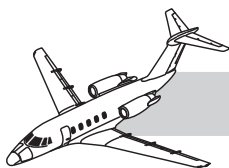


Figure 3-1. Cockpit Lighting Control Panels Utilized



Figure 3-2. Cockpit Floodlights



- Light control panel
- Environmental control panel
- Landing gear control panel
- Each throttle pedestal control panel

Map Lights

Map lights are on the left and right forward overhead panel with brightness controlled by a MAP LIGHT rheostat on the pilot and copilot side panels (Figure 3-3). All pilot map light rheostats are located on the aft portion of the CB panel and the copilot map light rheostats can be located either on the forward or aft portion of the CB panel depending on serial number and APU used.

INSTRUMENT LIGHTS

Instruments are illuminated either internally or by postlights. The panel lights are activated by the DAY/NITE DIM ON (Figure 3-1) switch on the left side PANEL LIGHT CONTROL subpanel, with illumination intensity controlled by the outer concentric LH rheostat. The illumination intensity of the center panel lights is controlled by the CTR rheostat.

The right instrument panel lights illumination intensity is controlled by the outer concentric right rheostat.

All panel lights are activated by the DAY/NITE DIM switch. The DAY/NITE DIM switch also dims the annunciator panel lights, gear lights, and engine instrument digital readouts.

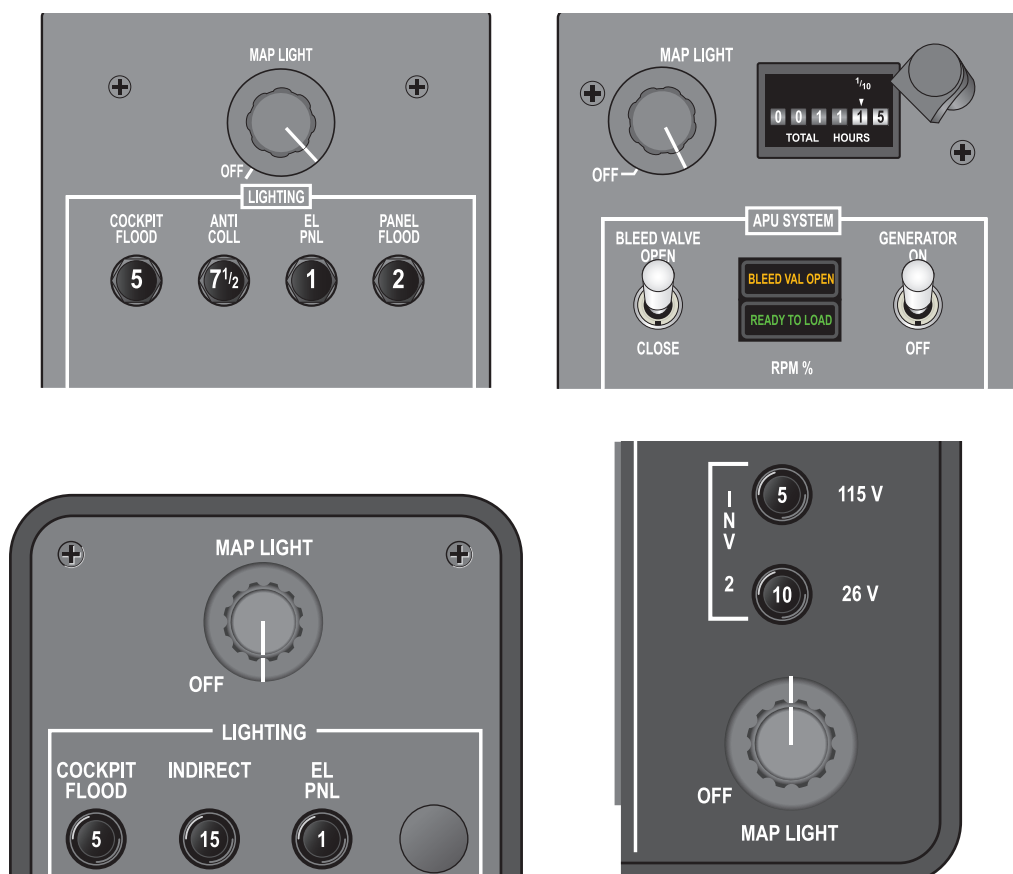


Figure 3-3. Map Light Rheostats

**Figure 3-4. Passenger Reading Lights****Figure 3-5. Cabin Entry Light Switch**

CABIN LIGHTING

Cabin lighting includes indirect fluorescent lighting, individual passenger reading lights, recessed footwell lights, and passenger information signs.

Passenger reading lights (Figure 3-4) are above each passenger seat. Each light is activated on or off with an adjacent pushbutton. The reading lights require normal DC power. Two reading lights on the right side serve as cabin entry lights and illuminate whenever the CABIN ENTRY light switch is on (Figure 3-5). The same two lights, plus the forward and aft reading lights on the left side, serve as emergency lights when the emergency lighting system is activated, regardless of switch position.

Cabin entry lights consist of two passenger reading lights. One is directly opposite the main cabin entrance door and the other is above the emergency exit. These lights are controlled by the CABIN ENTRY switch forward and inside the main cabin entrance door. These lights are powered from the hot battery bus and remain illuminated whenever the switch is on.

Footwell lights (Figure 3-6) require normal DC power and are activated by the AISLE LIGHT or FOOTWELL switch next to the CABIN ENTRY light switch.

**Figure 3-6. Footwell Lights**



Indirect fluorescent illumination is provided for the aisle and window areas. Illumination intensity is controlled by the BRIGHT and DIM switches or rheostats, which also allow individual control of these lights. These lights require normal DC power.

Two sets of passenger advisory lights (Figure 3-7) are controlled by a three-position switch on the center switch panel in the cockpit. The SEAT BELT switch position illuminates the fasten seat belt signs in the cabin. The PASS SAFE position illuminates the no smoking and fasten seat belt advisory lights, as well as four passenger reading lights (two on each side) and two EXIT signs. The OFF position extinguishes all passenger advisory signs. Safety chimes operate in conjunction with the SEAT BELT and PASS SAFE positions.

EMERGENCY LIGHTING

Emergency lighting provides illumination in case of primary electrical power failure or abnormal conditions. Emergency lighting consists of:

- Two Battery Packs
- Inertia Switch
- Four Passenger Reading Lights
- Overhead Service Lights
- Footwell Lights
- Main Entry Door Light



Figure 3-7. Passenger Advisory Lights

- Cabin Escape Hatch Light
- Emergency Exit Signs
- Exterior Ground Illumination Lights

The emergency lights are normally powered by the aircraft main DC system, but can be powered by the two NiCad emergency battery packs. The emergency battery packs are trickle charged from the aircraft DC system when normal DC power is available.

The lights are controlled with a three-position switch on the bottom of the left instrument panel (Figures 3-1 and 3-9). The switch positions are OFF–ON–ARM. When the switch is OFF, no emergency lights are illuminated. With normal DC power on the aircraft and with the switch OFF, an amber light adjacent to the switch is illuminated as a reminder to place the switch to either ON or ARM.



Figure 3-8. Emergency Exit Light

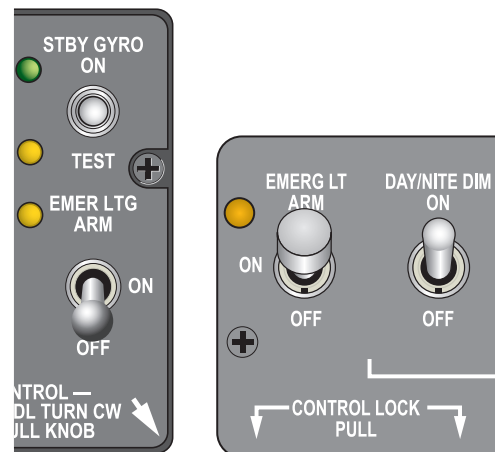


Figure 3-9. Emergency Light Switch Variations



When the switch is ON, the amber light adjacent to the switch extinguishes and all emergency lights are illuminated. The emergency lights are powered from the aircraft main DC system or from the battery packs if aircraft power is unavailable.

The switch is normally placed in ARM for flight. When the switch is in the ARM position, the amber light next to the switch extinguishes. The emergency lights, however, do not normally illuminate unless one of the following conditions exist:

- Passenger advisory switch is placed in PASS SAFE
- Normal DC power is lost
- 5g impact is sustained

Two of the three exterior emergency lights are located with the wing inspection lights on each side of the fuselage forward of the wings (Figure 3-10). The exterior emergency lights illuminate the wing area for night evacuation of the aircraft. A third emergency light, contained within a flush lens, is on the right side of the fuselage and illuminates the right wing walkway.

EXTERIOR LIGHTING

All exterior lights are controlled by switches on the instrument panel (Figure 3-11). Exterior lighting consists of:

- Navigation lights
- Anticollision (strobe) lights
- Wing inspection lights
- Taxi lights
- Recognition lights
- Ground recognition light (flashing)
- Tail (identification) floodlights
- Landing lights

NAVIGATION LIGHTS

The aircraft navigation lights consist of a green light in the right wingtip, a red light on the left, and a white light on the tip of each outboard flap island (Figure 3-12).

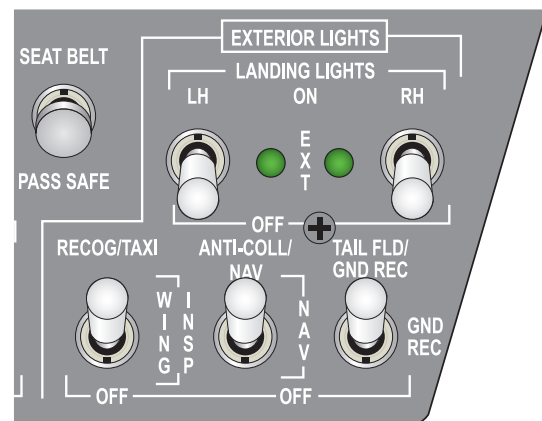


Figure 3-11. Exterior Lighting Controls

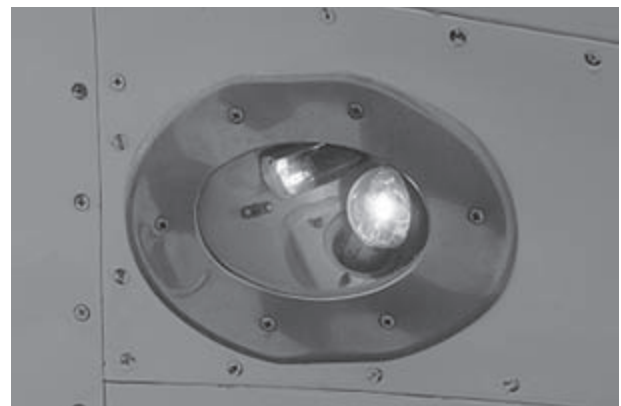


Figure 3-10. Emergency Walkway Lights

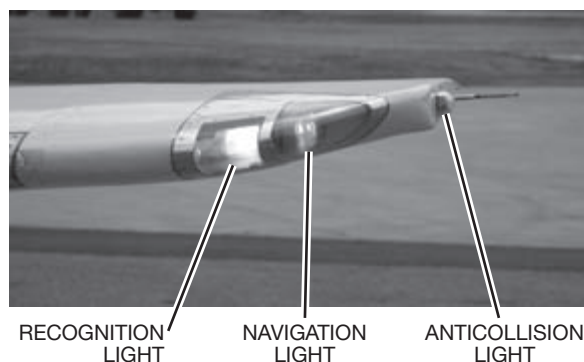
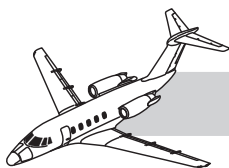


Figure 3-12. Navigation and Anticollision Lights

The navigation lights are controlled with an ANTI-COLL/NAV switch on the lower right corner of the EXTERIOR LIGHTS subpanel. In the NAV position, the navigation lights illuminate. The ANTI-COLL/NAV position causes simultaneous illumination of the navigation and the anticollision lights.

ANTICOLLISION LIGHTS

The anticollision (strobe) lights, which are in the wingtips (Figure 3-12) are extremely high intensity and can be disturbing to other pilots and ground personnel if used during ground operation. The anticollision lights are turned on immediately prior to takeoff roll and turned off shortly after landing. The lights are controlled with the ANTI-COLL/NAV switch.

WING INSPECTION LIGHTS

Wing inspection lights (Figure 3-13) illuminate the forward portion of the wing, enabling the pilot to detect ice buildup during night flights. The lights are controlled with the RECOG/TAXI switch on the EXTERIOR LIGHTS subpanel. The switch position WING INSP illuminates the lights. The wing inspection light assembly also houses an exterior emergency light. Refer to Emergency Lighting for the description and operation of the emergency lights.

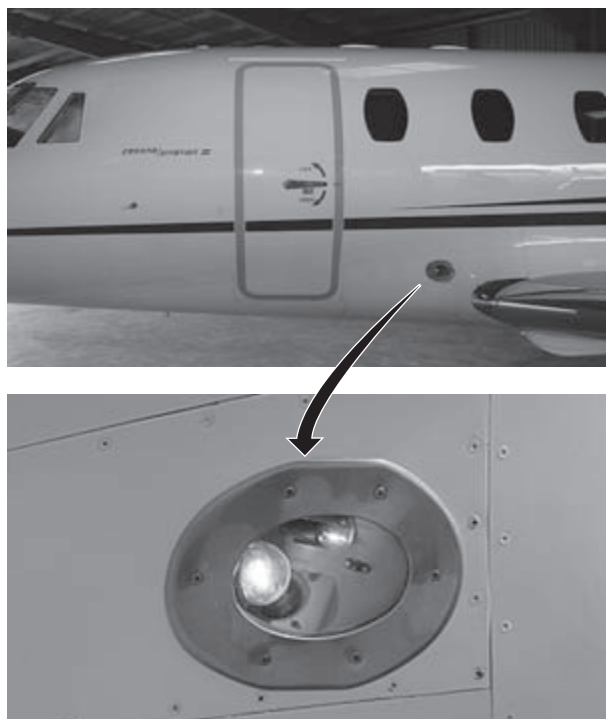


Figure 3-13. Wing Inspection Lights

TAXI LIGHTS

Taxi lights are on each main landing gear strut (Figure 3-14) and are controlled by the RECOG/TAXI-WING INSP-OFF switch during ground operation only. The landing gear squat switches preclude taxi light illumination in flight.

RECOGNITION LIGHTS

Recognition lights are mounted in the outboard forward edges of the wingtips (Figure 3-12). The RECOG/TAXI switch is used to activate these high-intensity lights.

GROUND RECOGNITION LIGHT

A flashing red ground recognition light (Figure 3-15) is on the bottom aft portion of the fuselage and is used when the aircraft is on the ground. An optional ground recognition light may be atop the horizontal stabilizer fairing. The lights are controlled with the TAIL FLD/GND REC switch on the center switch panel.



Figure 3-14. Taxi Lights



Figure 3-15. Ground Recognition Light

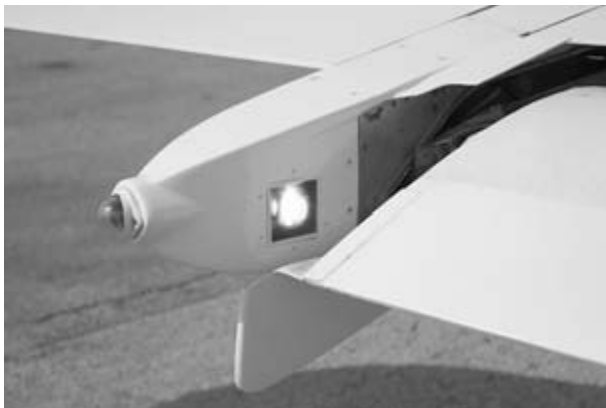


Figure 3-16. Tail Floodlight

TAIL FLOODLIGHTS

Some aircraft are equipped with optional tail floodlights that are used for additional illumination during night operations (Figure 3-16). The lights are controlled with the same switch used for the ground recognition light. This switch has an OFF, GND REC, and TAIL FLD/GND REC position. When the switch is

placed in the TAIL FLD/GND REC position, the lights illuminate. The lights are in the out-board flap islands and are aimed to illuminate the vertical stabilizer.

LANDING LIGHTS

Two retractable landing lights are on the left and right lower side of the nose (Figure 3-17). The lights are flush with the skin when retracted. The lights are controlled with the two switches on the LANDING LIGHTS subpanel. Each switch has three positions: LH-ON or RH-ON, EXT, and OFF. Placing either switch to the EXT position activates an internal motor in the landing light assembly and extends the associated light.

Two miniature green lights between the landing light switches remind the pilots that the landing lights are extended. Placing either switch to the ON position causes the associated light to extend and then illuminate. The lights illuminate only if fully extended. The maximum landing lights extended speed is 250 KIAS.

AFT BAGGAGE COMPARTMENT LIGHTING

A light in the baggage compartment provides interior lighting for baggage loading or unloading and for preflight purposes. A BAGGAGE LIGHT switch adjacent to the light powers a circuit wired through a microswitch on the forward door frame (Figure 3-18). Closing the access door extinguishes the light regard-



Figure 3-17. Landing Light



Figure 3-18. Baggage Compartment Light



Figure 3-19. Tailcone Compartment Light and Switch

less of the toggle switch position. The light is powered from the hot battery bus.

TAILCONE MAINTENANCE COMPARTMENT LIGHTING

The tailcone compartment light is on a swivel mount aft of the door frame (Figure 3-19).

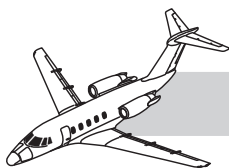
The TAILCONE LIGHT switch is mounted on the access door frame and is wired through the door-closed microswitch.

Closing the tailcone compartment door extinguishes the light regardless of the toggle switch position. This light can be detached from its mount and used as a hand-held light. It is powered from the hot battery bus.



QUESTIONS

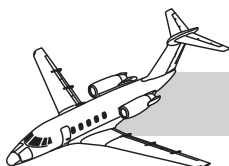
1. The lighting rheostat labeled “LH” controls the:
 - A. Pilot instrument panel lights
 - B. Center instrument panel lights
 - C. Copilot instrument panel lights
 - D. Both A and B
2. The lighting rheostat that controls the electroluminescent lighting is labeled:
 - A. LH
 - B. FD
 - C. CTR
 - D. EL
3. Placing the DAY/NITE DIM switch to ON:
 - A. Activates the control rheostats
 - B. Dims the annunciator panel lights
 - C. Illuminates the ice detection lights
 - D. All of the above
4. The map lights are controlled with rheostats on:
 - A. The center pedestal
 - B. The pilot and copilot instrument panels
 - C. The overhead lights panel
 - D. The pilot and copilot side panels



CHAPTER 4 MASTER WARNING SYSTEM

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The master warning system for the Citation 650 series aircraft warns the crew of aircraft equipment malfunctions, indications of unsafe operating conditions that require immediate attention, or indications that a particular system is in operation. Aural warnings also are used to draw attention to various situations.

The master warning system consists of two MASTER WARNING RESET switchlights and an annunciator panel. Each annunciator segment has a legend that illuminates to indicate an individual system fault or function. Red annunciators indicate a malfunction that

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DESCRIPTION

The master warning system and annunciator panel use a combination of aural and visual cockpit indications to advise the crew of important warnings, cautions, and advisory information about the aircraft and its systems.

Table 4-1 shows all annunciator combinations on the annunciator panel for the various models, their color, and cause for illumination.

Table 4-1. ANNUNCIATOR ILLUMINATION CAUSES

ANNUNCIATOR	CAUSE FOR ILLUMINATION	ANNUNCIATOR	CAUSE FOR ILLUMINATION
	Flashes when battery temperature exceeds 71°C or when BATT TEMP is selected by the rotary TEST knob. Steady illumination indicates that the battery temperature is between 60°C and 71°C. Triggers MASTER WARNING RESET switchlights.		Primary trim control or trim actuator is inoperative. Triggers MASTER WARNING RESET switchlights. (VII)
	Windshield heat on, temperature is excessive, system is shut down. Windshield bleed air on, temperature too hot, system is shut down. Illumination triggers the MASTER WARNING RESET switchlights.		Primary trim control or trim actuator is inoperative. Triggers MASTER WARNING RESET switchlights. (III & VI)
	Automatic emergency descent is initiated by the autopilot, if engaged, and the aircraft is above 34,275 feet with the cabin altitude above 13,500 feet. Triggers MASTER WARNING RESET switchlights. SNs 0001-0178 (III)		Smoke is detected in the aft cabin area. Triggers MASTER WARNING RESET switchlights.
	With the system in use, the windshield bleed air temperature is too hot. Bleed air to the windshield is shut off. Triggers MASTER WARNING RESET switchlights.		Middle two spoiler segments on each wing are not fully stowed.
	With the system in use, the windshield bleed air temperature is too hot. Bleed air to the windshield is shut off. Triggers MASTER WARNING RESET switchlights. SNs 0179-Sub (III) & VI		Fuel control computer has failed and/or been switched to manual control.
			Hydraulic firewall shutoff valves are closed by the LH or RH ENG FIRE PUSH switchlight.

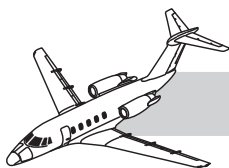


Table 4-1. ANNUNCIATOR ILLUMINATION CAUSES (Cont.)

ANNUNCIATOR	CAUSE FOR ILLUMINATION	ANNUNCIATOR	CAUSE FOR ILLUMINATION
FUEL LOW PRESS LH RH	Low fuel pressure in the engine supply line as indicated by a fuel pressure switch.	FUS TANK XFER FAIL WING FUEL XFER OPEN	Wing fuel transfer valve is energized open.
FUEL BOOST ON LH RH	Fuel boost pump has been turned on by either the low fuel pressure switch, the manual boost pump switch, start switch, or wing fuel transfer switch.	GEN OFF LH RH	Generator power relay is open. Illumination of both LH and RH triggers MASTER WARNING RESET switchlights.
FUS TANK LOW FULL	Indicates low or full fuselage tank as determined by high and low level float switches. <i>(III & VI)</i>	INVERTER FAIL 1 2	Inverter does not have a voltage output. Loss of both inverters triggers the MASTER WARNING RESET switchlights.
FUS TANK LOW FUS TANK FILL VLV	Indicates low fuselage tank as determined by a low level float switch. <i>(VII)</i>	STALL WARN ANTI SKID	The angle-of-attack system is inoperative and the stick shakers do not work.
FUS TANK LOW FUS TANK FILL VLV	Fuselage tank fill valve is not fully closed. <i>(VII)</i>	STALL WARN ANTI SKID	Indicates failure of the antiskid system or that the antiskid switch is OFF.
FUS TANK FILL VLV	Fuselage tank fill valve is not fully closed.	STAB ANTI-ICE LH RH	Horizontal stabilizer ice protection system is inoperative because of overheating, loss of power, or heater failure.
FUS TANK XFER FAIL WING FUEL XFER OPEN	Fuselage fuel transfer pump or pumps have failed, or the fuselage transfer valve failed to open.	ENG ANTI-ICE LH RH	Engine inlet anti-ice protection system is inoperative because of inadequate bleed air heat, wing fairing heater failure or overheat, or failure of the pressure/temperature sensor.

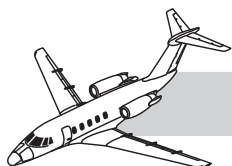


Table 4-1. ANNUNCIATOR ILLUMINATION CAUSES (Cont.)

ANNUNCIATOR	CAUSE FOR ILLUMINATION	ANNUNCIATOR	CAUSE FOR ILLUMINATION
	Wing anti-ice system is inoperative because of overheating or inadequate bleed air temperature.		a. On steady in flight indicates one of the four door pin switches or the door handle is not in the locked position. b. Citation III/VI—steady on the ground, door is open. c. Flashing on the ground indicates one of the four door pin switches or the door handle is not in the locked position.
	Emergency pressurization turned on by either the 13,500-foot barometric switch or by the ENG BLD AIR switch.		Left nose equipment door, baggage compartment door, toilet service door, and/or single point pressure refueling door is not locked. (Depending on serial number or service bulletin.)
	High pressure bleed air to the environmental control unit was turned on automatically or manually.		Oil pressure is less than 25 psi. Illumination of the annunciator triggers the MASTER WARNING RESET switchlights.
	The isolation valve between the cockpit and cabin environmental control unit is energized open.		(VII)
	The CKPT PAC and/or CAB PAC switch is positioned to HIGH, causing the flow control valve to be in high flow.		Cabin altitude is greater than 8,500 feet. (III & VI)
	Indicates a failure in the fire detection system.		Secondary trim is on, but the clutch is not engaged (actuation of trim switches should engage clutch), or secondary trim is turned off but the clutch has not disengaged.
	Indicates a loss of bleed air pressure to the primary cabin door seal.		Improper takeoff configuration of spoilers, speedbrakes, flaps, stabilizer trim, gust lock, APU operation if ground only, or unlocked door.

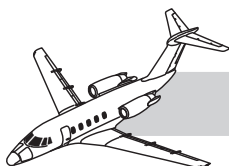


Table 4-1. ANNUNCIATOR ILLUMINATION CAUSES (Cont.)

ANNUNCIATOR	CAUSE FOR ILLUMINATION	ANNUNCIATOR	CAUSE FOR ILLUMINATION
SPOILERS UP SPOILER HOLDOWN	The hold down return line is pressurized, spoilers/speedbrakes cannot be extended. Roll spoilers are operable if auxiliary hydraulic pump is operating	SPOILERS UP SPOILER HOLDOWN	The inboard spoiler segments are not fully stowed.
FUEL LOW LEVEL LH RH	The amber fuel low level light indicates that wing fuel in the wing is less than approximately 350 pounds.	HYD PRESS LOW LH RH	Hydraulic pressure switches indicate low pressure output from the engine-driven pumps.
RUDDER BIAS AIL BOOST OFF	The solenoid bypass valve is open, reducing the effectiveness of the rudder assist cylinder.	HYD VOL LOW AUX HYD PRESS	Fluid level at or below 150 cubic inches as indicated by a microswitch activated mechanically by the reservoir.
RUDDER BIAS AIL BOOST OFF	The aileron boost pressure has been lost.	HYD VOL LOW AUX HYD PRESS	Auxiliary hydraulic pump is producing pressure.
FUEL F/W SHUTOFF LH RH	The fuel firewall shutoff valve is closed by the LH or RH ENG FIRE PUSH switchlight.	HYD VOL LOW AUX-HYD PUMP ON	Auxiliary hydraulic pump is producing pressure.
FUEL FLTR BYPASS LH RH	Fuel filters are approaching or are actually being bypassed because of filter contamination.	GUST LOCK GROUND IDLE	The control surface and throttle lock are not fully disengaged.
FUS TANK FUEL PUMP 1 2	Fuselage tank fuel pumps are being powered.	GUST LOCK GROUND IDLE	Indicates that the engines are configured for reduced idle. The annunciator should be illuminated for all ground operations. On landing, it should illuminate 8 seconds after touchdown with the switch in NORM.

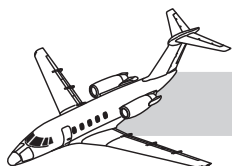


Table 4-1. ANNUNCIATOR ILLUMINATION CAUSES (Cont.)

ANNUNCIATOR	CAUSE FOR ILLUMINATION	ANNUNCIATOR	CAUSE FOR ILLUMINATION
	The cockpit electric auxiliary heater has overheated and has shut down. (III & VI)		The environmental control unit is disabled because of an overheat condition in the compressor or turbine.
	The baggage compartment heater has overheated. (III & VI)		Temperature in the cockpit or cabin duct is excessive.
	Depending on service bulletin, with the system in use the windshield bleed air temperature is too hot or too cold. With the system off, the light indicates trapped pressure in the air duct.		Pressure switches indicate low pressure in one or both of the engine fire extinguisher bottles. (VII)
	The parking brake is not fully disengaged. (III & VI)		Pressure switches indicate low pressure in one or both of the engine fire extinguisher bottles. (III & VI)
	The amber windshield or temperature sensor fault light indicates that the electric windshield controller has failed or that a phase imbalance has been detected in the system. (VII)		Magnetic contacts have detected a metal chip in the oil pump discharge back to the oil tank.
	Current-sensing relays indicate one or more of the pitot or static anti-ice heaters are inoperable or the switch is off.		A current limiter or circuit breaker in the aft power junction box is open.
	High pressure bleed air from precooler has exceeded 550°F.		



CONTROLS AND INDICATIONS

ROTARY TEST KNOB

All annunciator bulbs can be tested by placing the rotary TEST knob (Figure 4-1) on the center switch panel to the ANNUN position. This action illuminates all annunciators and causes the MASTER WARNING RESET switchlights to flash. Illumination verifies only annunciator lamp integrity. Other annunciators illuminate when the rotary TEST knob is used to test other systems. Burned out annunciator bulbs can be quickly replaced from the front of the panel by pushing the lens in, pulling it out, and rotating 90° to expose the base. After replacing the bulb, reseal the lens and press in until it latches.

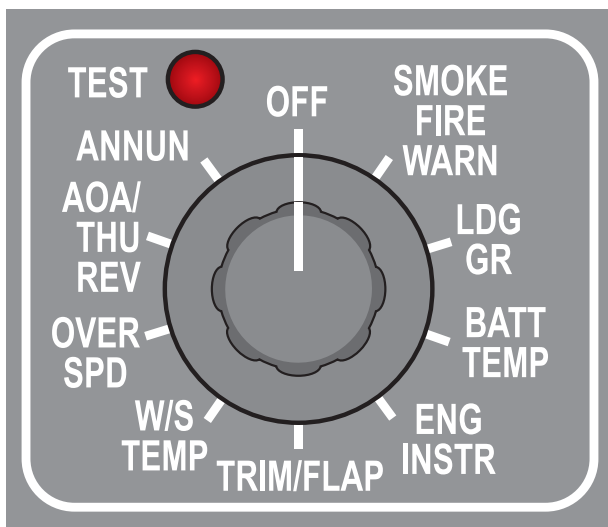


Figure 4-1. Rotary TEST Knob

ANNUNCIATOR PANEL

The annunciator panel is on the upper center instrument panel. Each annunciator capsule contains a legend pertinent to its function. The annunciators are arranged according to aircraft systems. The annunciator system is powered from the left extension and right branch DC buses through the WARN LTS 1 and 2 circuit breakers on the pilot CB panel.

When a system malfunctions, the associated annunciator illuminates and remains illuminated until the malfunction is corrected. Some annunciators illuminate indicating normal system operation and extinguish when system operation is terminated.

MASTER WARNING RESET SWITCHLIGHTS

There are two MASTER WARNING RESET switchlights, one each on the pilot and copilot instrument panels. When any red annunciator illuminates, the MASTER WARNING RESET switchlights illuminate simultaneously and flash until reset. There are three conditions during which amber annunciators cause the MASTER WARNING RESET switchlights to illuminate:

- LH and RH GEN OFF annunciators illuminate
- 1 and 2 INVERTER FAIL annunciators illuminate
- The thrust reverser ARM or UNLOCK annunciators illuminate in flight

The MASTER WARNING RESET switchlights can be reset by pressing either switchlight. Resetting the MASTER WARNING RESET switchlights rearms the system so it functions with the illumination of another annunciator. The MASTER WARNING RESET switchlights flash during the ANNUN test mode but cannot be reset. A loss of either left or right bus power to the annunciator panel causes a MASTER WARNING RESET switchlight to illuminate steady. Loss of power through the WARN LTS 1 circuit breaker causes the left MASTER WARNING RESET switchlight to illuminate steady. The switchlight cannot be reset. Loss of power through the WARN LTS 2 circuit breaker causes the right MASTER WARNING RESET switchlight to illuminate steady. The switchlight cannot be reset. The MASTER WARNING RESET switchlights do not dim during night operation.



AURAL WARNINGS

Various aural warnings are incorporated into aircraft systems to warn of specific conditions and malfunctions.

The aural warning system can be tested using the same rotary TEST knob used for testing the annunciator system (Table 4-2). When the knob is rotated through each position, the associated warning functions occur.



CITATION 650 SERIES PILOT TRAINING MANUAL

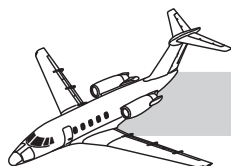
Table 4-2. TEST INDICATIONS

SWITCH POSITION	INDICATION
OFF	The red TEST indicator is extinguished and the test system is inoperative.
SMOKE FIRE WARN	The SMOKE DETECT annunciator and MASTER WARNING RESET switchlights illuminate. The fire bell sounds for approximately 3 seconds, the LH and RH FIRE PUSH switchlights illuminate, and the LH and RH FIRE DET FAIL annunciator illuminates.
LDG GR	The three green safe lights and the red unlocked light on the landing gear control panel illuminate and the warning horn sounds. The warning horn cannot be silenced with the flaps extended beyond 20°.
BATT TEMP	The BATT O'TEMP 1-2 annunciators alternately flash and the MASTER WARNING RESET switchlights illuminate. The optional battery temperature LED indicator indicates -188°, and the red and amber lights illuminate on the indicator.
ENG INSTR	The left and right igniter lights in the interturbine temperature indicator illuminate. The digital N ₂ turbine speed indicator reads 88.8 and the red and green lights illuminate. The fuel temperature reads -88°C and 8s appear in the optional fuel quantity totalizer gauge.
TRIM/ FLAP	The PRI TRIM FAIL annunciator and MASTER WARNING RESET switchlights illuminate when the primary trim switches are actuated. The FLAPS INOP light near the flap control handle illuminates and extinguishes without being reset. The "no take off" warning horn sounds. The FLAP O'SPD lights illuminate, if installed.
W/S TEMP	The red W/S OVHT and MASTER WARNING RESET switchlights illuminate. On aircraft without the W/S OVHT light, only the amber W/S AIR light illuminates. The windshields for SNs 7001 – 7119 are electrically heated. Placing the W/S BLEED switch ON will illuminate the red W/S O'HEAT LH-RH and MASTER WARNING RESET switchlights. The red W/S O'HEAT LH-RH, MASTER WARNING RESET switchlights and amber W/S FAULT LH-RH lights illuminate for a few seconds and then extinguish when the windshield anti-ice switches are turned on (after engine start).
OVER SPD	The V_{MO}/M_{MO} warning horns sound. Satisfactory operation of both horns is detected by the beat frequency. The optional TAS/TAT/SAT indicator shows: TAT -16°C, SAT -45°C, TAS 466 kts. NOTE: The avionics power switch must be on to check the TAS/TAT/SAT indicator, which checks the overspeed warning horns.
AOA/ THU REV	The FLAPS SPDBK/SP and AOA PROBE lights near the angle-of-attack indicator illuminate and the STALL WARN annunciator illuminates. The angle-of-attack indicator flag and the flight director fast/slow OFF flags appear. The angle-of-attack indicator pointer slows to approximately "0" scale and pauses. After approximately 40 seconds, the FLAPS SPDBK/SP, AOA PROBE, and STALL WARN annunciators extinguish and the angle-of-attack indicator and fast/slow OFF flags clear. The fast/slow indicators indicate FAST and the indicator pointer moves up scale. The stick shakers operate when the angle-of-attack indicator pointer indicates between 0.82 ± 0.02 on the scale. The angle-of-attack pointer continues to 1.0, the fast/slow indicators indicate SLOW and the indicator OFF flags appears. The OFF flag clears and the pointer returns through zero to its position before the test. AOA indexer(s) on the glareshield illuminate in sequence with AOA indicator pointer. The left and right thrust reverser ARM, UNLOCK, and DEPLOY lights and the MASTER WARNING RESET switchlights illuminate.
ANNUN	All the annunciators and the MASTER WARNING RESET switchlights illuminate. When avionics power switch is on, the altitude alert horn sounds and the altitude alert lights illuminate. The MASTER WARNING RESET switchlights cannot be cancelled. The warning lights above the electronic attitude director indicator illuminate along with various other lights associated with the specific avionics system installed.



QUESTIONS

1. An annunciator extinguishes:
 - A. When pressed
 - B. Upon landing
 - C. When the malfunction is corrected or when the system operation is terminated or complete
 - D. If the MASTER WARNING RESET switchlights are reset
2. The MASTER WARNING RESET switchlights illuminate:
 - A. When any annunciator illuminates
 - B. When a red annunciator illuminates
 - C. When the LH and RH GEN OFF annunciators illuminate
 - D. Both B and C
3. The rotary TEST knob:
 - A. In the ANNUN position, illuminates all annunciators
 - B. Is spring-loaded to OFF
 - C. In the ANNUN position, illuminates only red annunciators
 - D. In the ANNUN position, illuminates only amber annunciators

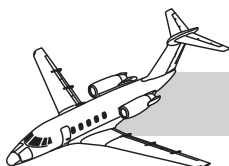


CHAPTER 5

FUEL SYSTEM

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CHAPTER 5

FUEL SYSTEM



INTRODUCTION

The Citation 650 series aircraft fuel system provides an independent fuel system for each engine with fuel transfer capability. The fuselage tank can be used for additional fuel storage, and the system can be serviced by either a single-point or overwing method.

GENERAL

The aircraft fuel system consists of identical left and right wing tanks and a fuselage tank. The wing fuel tanks are sealed wing structures (wet wings), while the fuselage tank consists of a bladder inside a sealed metal tank. Each wing fuel tank supplies fuel to its respective engine via an engine feed reservoir in the inboard center section of each wing tank. The reservoir is normally supplied with fuel from

the wing area by scavenge ejector pumps. Fuel can be transferred from wing to wing via the feed reservoirs to correct any fuel imbalance. The fuselage tank is transferred directly into both wings equally. Fuel transfer is controlled with boost pump switches, a fuselage tank transfer switch, and a wing transfer switch. Fuel transfer is monitored via the fuel quantity gauges and annunciator lights.



DESCRIPTION

The fuel system is composed of a fuel storage system and a fuel distribution system.

Fuel Storage

Wing fuel storage consists of the forward fairing area, the wing area, and the engine feed reservoir (Figure 5-1). Each wing tank is a sealed wing structure containing the fuel vent system, baffles, flapper valves, transfer tubes, and scavenge ejector pumps. The fuselage tank is a bladder within a sealed metal tank.

Each wing has a capacity of 3,242 pounds of usable fuel, while the fuselage tank has a capacity of approximately 996 pounds usable fuel for a total of 7,480 pounds (approximately 1,109 U.S. gallons).

Fuel Distribution

Fuel distribution involves electric boost pumps, ejector pumps, firewall shutoff valves, fuel pressure switches, a fuel filter (part of the engine-driven fuel pump), and fuel lines. These components ensure a constant supply of fuel to the engine-driven fuel pumps. The fuel system supplies each engine with fuel from its respective side of the aircraft. Fuel may be transferred between engine feed reservoirs to correct a fuel imbalance, or if an engine fails, to allow transfer of all fuel to the feed reservoir of the operating engine.

Normal fuel flow is from the wingtip to the wing root through flapper valves. Reverse flow through the wing is through the transfer tubes for single-point refueling. The forward fairing area feeds into the main wing area via a scavenge ejector pump. If a scavenge ejector pump fails, fuel can gravity-feed to the main wing area through flapper valves.

Reverse flow from the wing to the forward fairing occurs through a standpipe until the fuel level drops below the standpipe. At that point, the remaining fuel in the forward fairing tank is pumped into the wing. At the wing root, fuel is pumped into the engine feed reser-

voir by two scavenge ejector pumps, or it can be gravity-fed through flapper valves.

The engine feed reservoir contains the electric boost pump and the primary ejector pump, which provide fuel pressure to the engine-driven fuel pump and the scavenge ejector pumps.

COMPONENTS

Fuel Tank Relief Valves

The fuel tanks are vented through internally routed lines and a vent near the wingtip (Figure 5-2). The vent includes a vent float valve and an overboard opening outboard of the aileron. Relief valves within each wing relieve positive or negative pressure inside the wing if a vent or the pressure refueling system malfunctions (Figure 5-2). The positive pressure relief valve opens at a positive pressure of 5 psig and resets when the pressure decreases to 1.25 psig. The negative pressure relief valve opens at about 0.5 psig negative pressure to allow ambient pressure into the wing. The fuselage tank is individually vented through a line shrouded by a drain mast on the bottom of the fuselage (Figure 5-3).

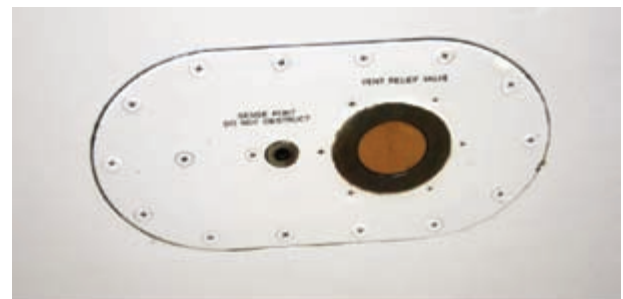


Figure 5-2. Main Tank Vent and Relief Valves

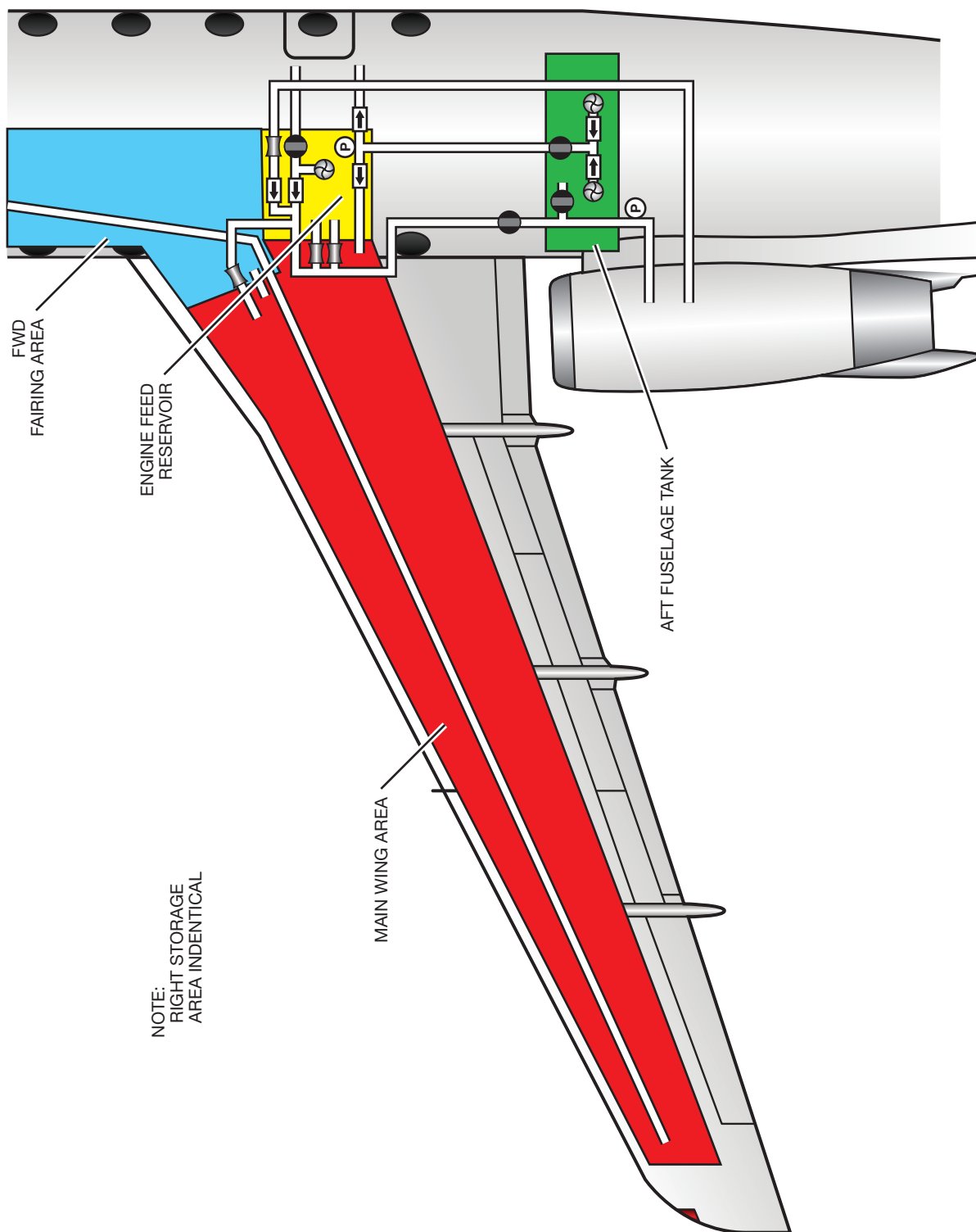
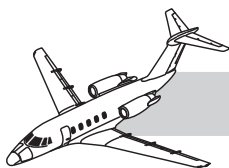


Figure 5-1. Fuel Tank Schematic



**WING FUEL TANK
DRAIN VALVES**



**FUSELAGE FUEL TANK
MAST AND DRAIN VALVES**

Figure 5-3. Fuel Tank Drain Valves and Fuselage Drain Mast

Fuel Tank Drains

Nine poppet drain valves are on the lower surface of the center wing—five on the left and four on the right. Three additional drain valves are under the fuselage tank. The drain valves are opened by an Allen (hex) tool for draining moisture, sediment, or fuel from the tank (Figure 5-3).

Electric Fuel Boost Pump

The electric FUEL BOOST pump in the feed reservoir provides fuel pressure for engine starting and fuel transfer and backup pressure for the primary ejector pump. Each pump has a three-position FUEL BOOST pump switch on the center switch panel (tilt panel) (Figure 5-4).

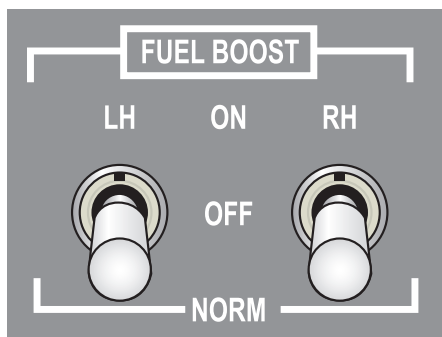


Figure 5-4. FUEL BOOST Pump Switches

Ejector Pumps

Primary Ejector Pump

The primary ejector pump in each engine feed reservoir is the primary source of pressurized fuel for the engine-driven fuel pump (Figure 5-5). The primary ejector pump uses motive flow from the engine-driven fuel pump to pick up fuel from the engine feed reservoir and deliver it to the engine-driven fuel pump. The high pressure fuel creates a venturi effect and develops a low pressure (suction) at the pump inlet (Figure 5-6). The suction draws a large volume of fuel from the engine feed reservoir into the supply line at a lower pressure. The pump operates whenever the engine is running, and since it has no moving parts, it is extremely reliable.

Scavenge Ejector Pumps

A scavenge ejector pump in the forward fairing tank is a continuously operating ejector pump, which uses (as motive flow) pressurized fuel from the primary ejector pump or the electric boost pump. The scavenge ejector pump transfers fuel from the forward fairing tank into the wing tank. Fuel returns from the wing tank to the forward fairing tank through a standpipe. When the fuel level drops below the standpipe, the scavenge pump transfers the remaining fuel in the forward fairing area to the main wing. If the scavenge pump fails, the fairing tank fuel can enter the wing tank through flapper valves.

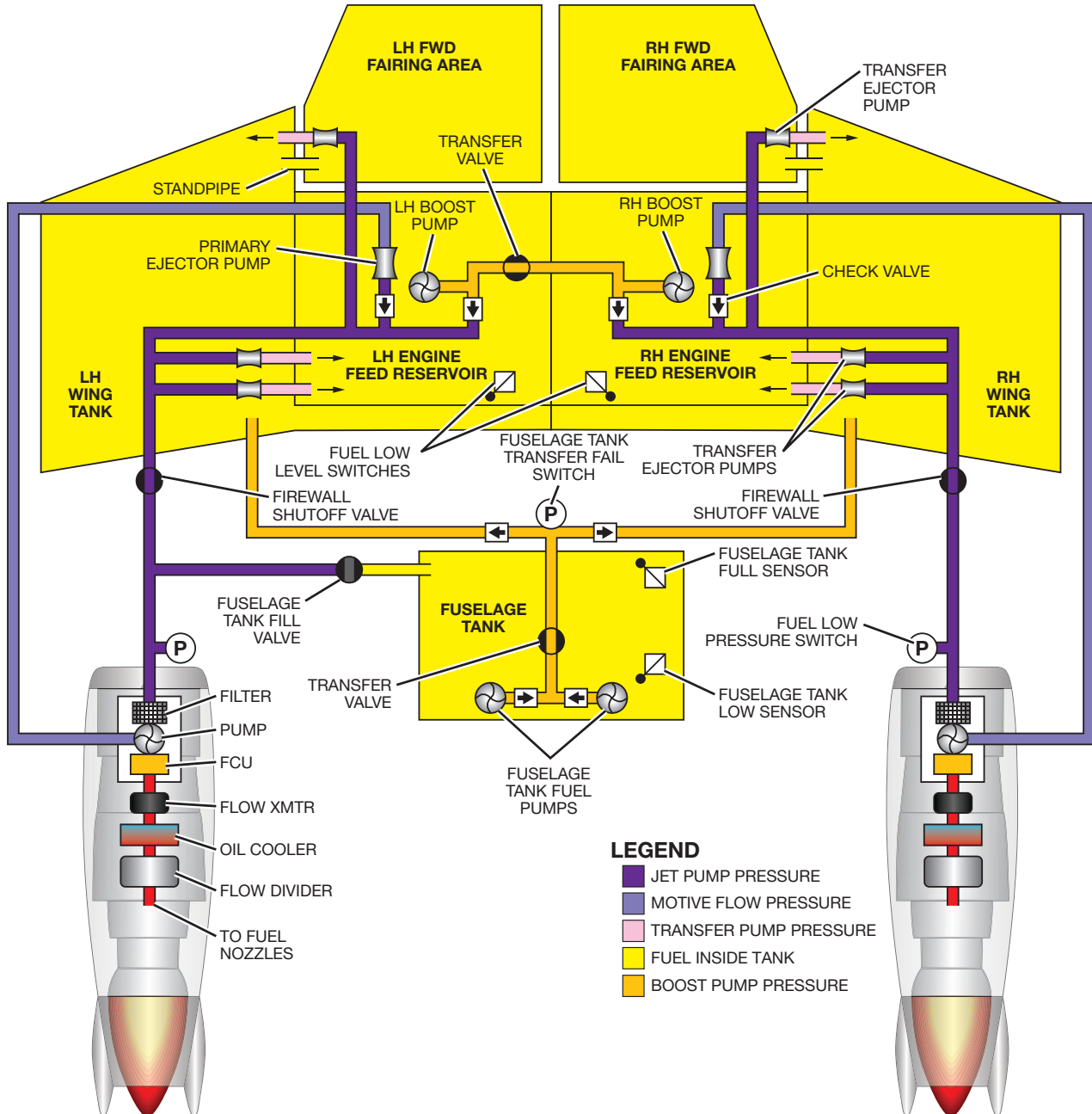


Figure 5-5. Fuel Distribution

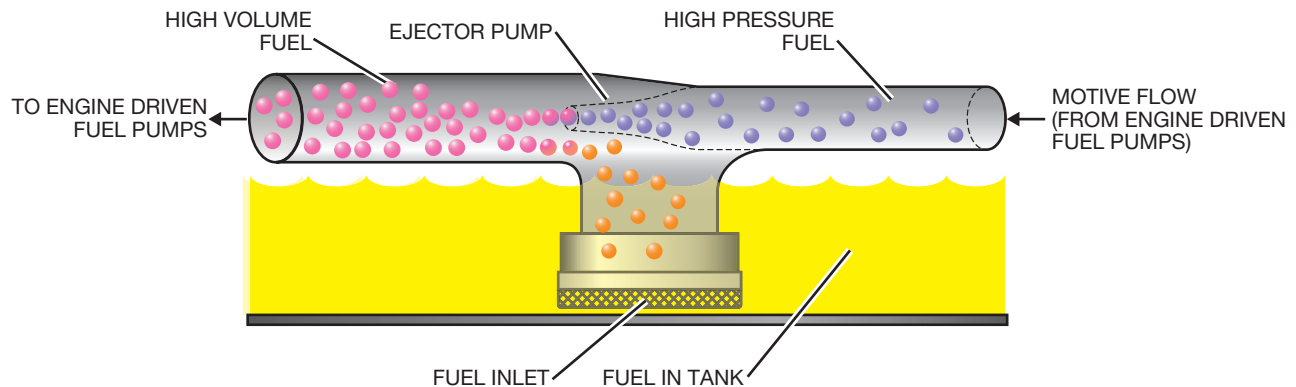


Figure 5-6. Ejector Pump

Two scavenge ejector pumps are in the wing tank, where the wing tank joins the engine feed reservoir. The scavenge ejector pumps use pressurized fuel from the primary ejector pump or the electric boost pump as motive flow. The pumps transfer fuel from the wing tank to the engine feed reservoir. If either pump fails, the fuel can feed into the engine feed reservoir through flapper valves.

Firewall Shutoff Valves

A firewall shutoff valve is in each fuel supply line aft of the aft pressure bulkhead. Firewall shutoff valves are motor-operated by 28 VDC from the emergency DC bus (SNs 0152 and subsequent) and are closed by the LH–RH ENG FIRE PUSH switchlights (Figure 5-7).

LH–RH ENG FIRE PUSH switchlights are above the forward annunciator panel. If an engine fire is detected, the red LH or RH ENG

FIRE PUSH switchlight illuminates. The switchlight closes its respective firewall shutoff valve, terminating fuel flow to the engine.

Fuel Pressure Switch

A fuel pressure switch in each engine fuel supply line opens and closes at predetermined settings. When the switch closes at low pressure, the FUEL LOW PRESS LH–RH annunciator illuminates. If the FUEL BOOST pump switch is in NORM, the boost pump activates automatically.

CONTROLS AND INDICATIONS

Fuel Quantity Indicator

Fuel quantity is indicated by a capacitance system. The left and right wing tanks and the fuselage tank are gauged separately. The amount of fuel in each tank is displayed on the fuel quantity indicator (Figure 5-8). The fuel amount is determined by 32 probes, 15 in each wing tank and two in the fuselage tank. The optional, multi-function fuel totalizer has three, 3-position toggle switches spring-loaded to the center position (Figure 5-8 and 5-9).

Fuel Temperature Indicator

A dual digital fuel temperature indicator on the center instrument panel displays fuel temperature in both engine feed reservoirs (Figure 5-10).

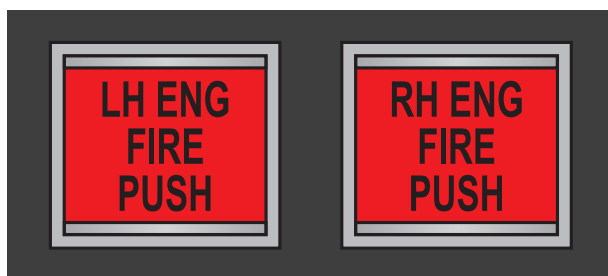


Figure 5-7. LH–RH ENG FIRE PUSH Switchlights

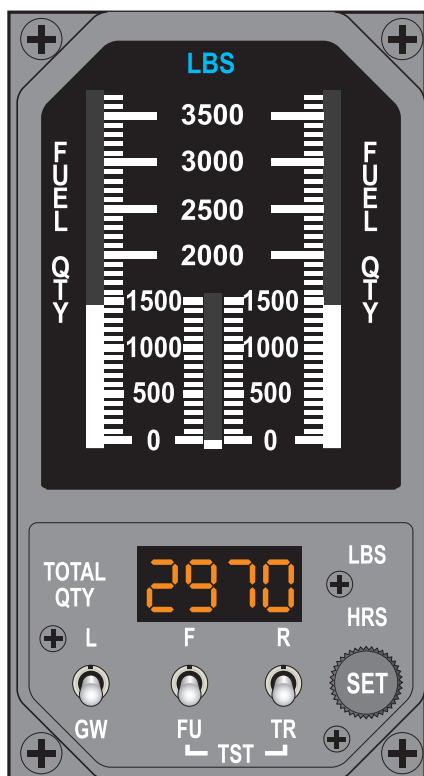
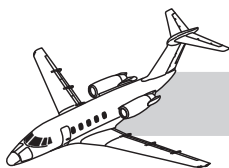


Figure 5-8. Fuel Quantity Indicator with Optional Totalizer

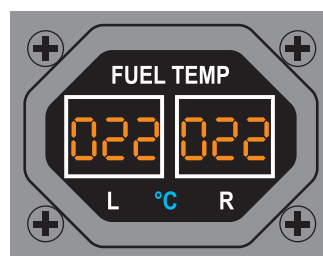


Figure 5-10. Fuel Temperature Indicator

FUEL XFER Switch

The WING FUEL XFER switch controls the fuel transfer valve, which permits fuel transfer between engine feed reservoirs (Figure 5-11).

Fuel Boost Pump Switches

The LH or RH FUEL BOOST switch positions are ON–OFF–NORM (Figure 5-4). In the ON position, the pump operates continuously. In the OFF position, the pump is deenergized, except when the start button is pressed or fuel transfer from the respective tank is selected. The NORM position is the usual operating mode. The pump activates in this position during engine start, wing fuel transfer or when



NOTE 1: GROSS WEIGHT IS ADJUSTED BY THE SET KNOB. TOTALIZER REDUCES GROSS WEIGHT AS FUEL IS USED.
NOTE 2: IF POWER IS INTERRUPTED TO THE TOTALIZER, GROSS WEIGHT READOUT RESETS TO 15,000 POUNDS.
NOTE 3: FUEL USED READOUT IS RESET TO ZERO WHEN POWER IS APPLIED TO THE TOTALIZER.
NOTE 4: TIME REMAINING BASED UPON PRESENT FUEL FLOW.

Figure 5-9. Fuel Quantity Totalizer Functions

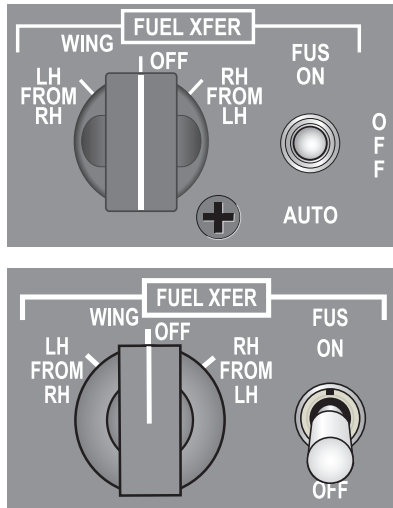


Figure 5-11. FUEL XFER Switch Variations

the low pressure switch in the supply line closes because of low fuel pressure. During engine shutdown, the low pressure switch is removed from the circuit when the throttle is in the CUT OFF position, thus preventing activation of the electric boost pump as fuel pressure drops.

Fuel System Annunciators



FUEL F/W SHUTOFF LH-RH—Illuminates when the respective ENG FIRE PUSH switchlight is pressed, indicating the fuel fire-wall shutoff valve is closed.



FUS TANK XFER FAIL—Illuminates to indicate low fuel pressure in the transfer manifold. Illumination is inhibited for 8 seconds after transfer pumps are activated to allow pressure to increase. If one or both pumps fail or if pressure drops, the annunciator illuminates. The combined pressure output of both pumps is required to prevent illumination of the annunciator. Fuel transfer to the wings can occur on one pump and is verified by observing the fuselage fuel quantity gauge.



WING FUEL XFER OPEN—Illuminates when the fuel transfer solenoid valve is energized.



FUEL LOW LEVEL LH-RH—Illuminates when float switches in either wing tank detect that usable wing fuel quantity drops to 350 pounds or less. This system is completely independent of the fuel quantity gauges.



FUEL LOW PRESS LH-RH—Illuminates when fuel pressure in the respective engine fuel supply line drops below a pre-determined setting.



FUS TANK LOW-FULL—Illuminates to indicate the fuselage tank is either filled to capacity or nearly empty when transferring to the wings. (Only Citation III/VI have the FULL annunciator.)



FUS TANK FILL VLV—Illuminates when the fuselage tank fill valve is not fully closed.



FUEL FLTR BYPASS LH-RH—Illuminates to indicate the fuel filter is contaminated and being bypassed.



FUEL BOOST ON LH-RH—Illuminates to indicate the boost pumps have been manually selected to on, or automatically by low fuel pressure, fuel transfer, or during start cycle.



FUS TANK FUEL PUMP 1-2—Illuminates to indicate the fuselage tank fuel pumps are being powered.

OPERATION

The single-point pressure refueling (SPPR) system can refuel or defuel any combination of tanks (Figure 5-12). The SPPR panel is inside a door on the fuselage aft of the right



CITATION 650 SERIES PILOT TRAINING MANUAL

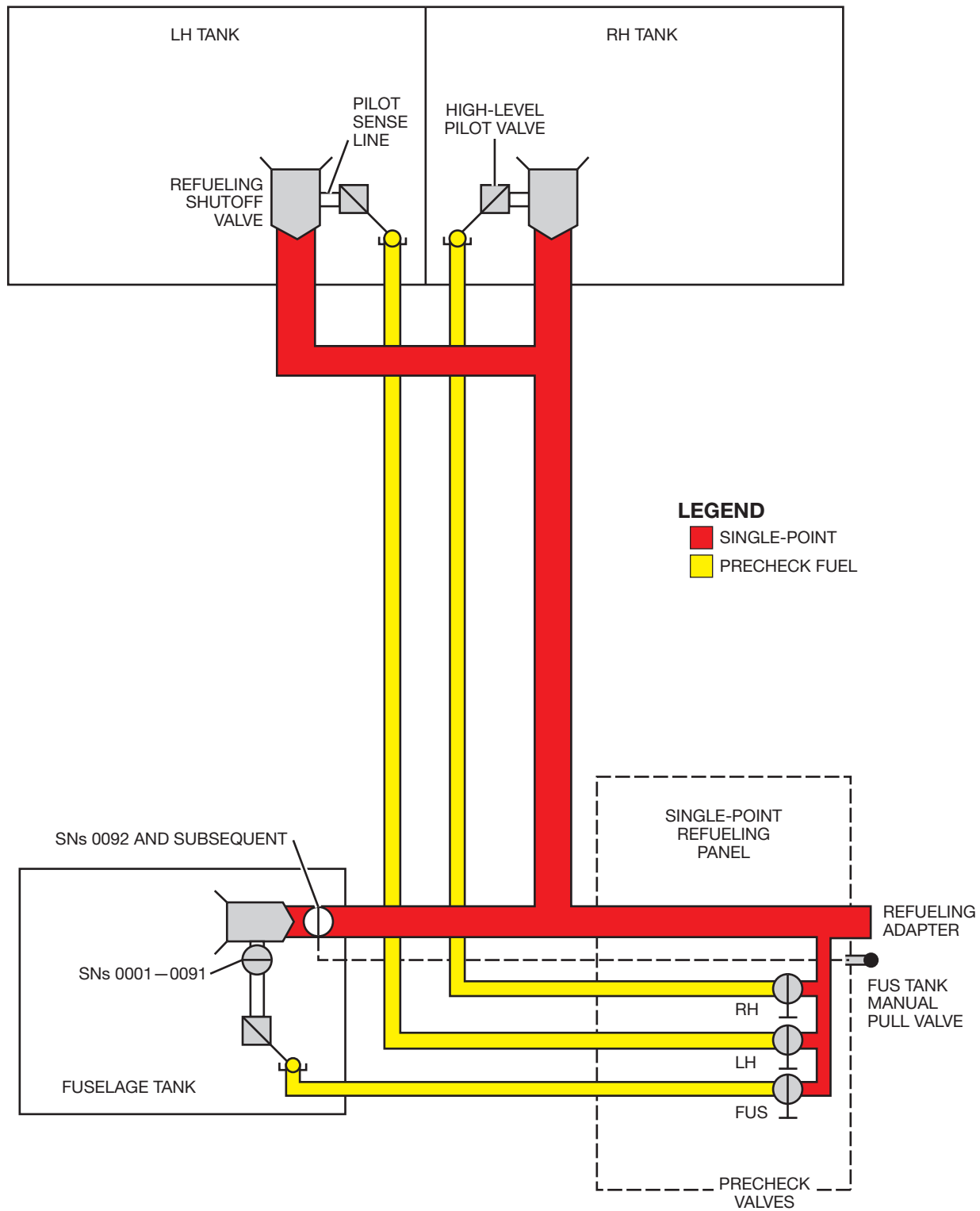


Figure 5-12. Single-Point Pressure Refueling System



wing. Refueling instructions are on the inside of the door (Figure 5-13). The system requires a minimum of 6 psi for refueling and cannot be damaged by pressure or suction. Electrical power is not required for the SPPR system.



Figure 5-13. SPPR Panel

PRECHECK

To use the SPPR method, remove the dust cover from the refueling adapter and connect the refueling nozzle. If the fuselage tank is being refueled, pull out the FUSE TANK FILL knob at the forward lower corner of the panel.

NOTE

The fuselage tank will not precheck or fill with the knob pushed in.

Start the flow of fuel, and then lift the toggle switch for each tank to be filled. The flow should stop within approximately 10 seconds but no longer than 30 seconds. If the system fails the precheck, terminate single-point refueling.

REFUELING

After the system has made a successful pre-check, place the toggles down to refuel.

When the tanks are full, the system shuts off the same as it did in the precheck. Turn off the fuel truck or hydrant pump, and then remove the hose from the aircraft. Replace the dust cover and close the refueling door.

NOTE

The door can not close unless all toggle valves are in the down position and the fuselage tank knob is pushed in.

For a fuel imbalance, compute the number of gallons to be loaded in each wing. After the precheck, leave one wing toggle up, and add the desired quantity to the other wing. Use the truck or hydrant meter to aid in determining the desired quantity. When the first tank is fueled to the desired level, repeat the steps for the other tank.

DEFUELING

The entire aircraft can be defueled by connecting the single-point nozzle and applying suction. To defuel the fuselage tank pull out the FUSE TANK FILL knob. One or both wing tanks may be prevented from defueling with two defuel valves ahead of the main landing wheel wells (Figure 5-14). To prevent a wing tank from defueling, open the door and pull the toggle down. When defueling is complete, push the toggle in and secure the access door.



Figure 5-14. Wing Defuel Valve



NOTE

The defuel valve access doors can not close if the toggle is not pushed in.

SNs 0001—0091

When suction is applied to the single point system, fuselage tank defueling cannot be prevented.

OVERWING REFUELING

Overwing refueling of both wing tanks uses conventional fuel truck nozzles. The overwing refueling method adds approximately 30 gallons more fuel than the single-point method. One fuel filler cap is on the upper surface of each wing near the wingtip (Figure 5-15). The fuselage tank does not have a filler.



Figure 5-15. Overwing Refueling

To refuel the fuselage tank, verify that the left wing tank contains a minimum of 175 gallons of fuel. Secure the cap and place the fuselage fill switch under the left wingtip to the FILL position (Figure 5-15). The FILL position activates the left boost pump and opens the fuselage fill valve, transferring fuel from the left engine feed manifold to the fuselage tank (Figure 5-6). The system, which services the fuselage tank, automatically shuts off the left boost pump and closes the fuselage fill valve when the tank is full via the high-level float switch. The fuselage fill switch must be turned OFF in order to close the access door. This system is powered from the hot battery bus.

NOTE

In all refueling operations, the aircraft and the refueling truck must be grounded, and the aircraft must be bonded to the fuel truck.

SNs 0001—0076 Not Modified by SB 650-28-20

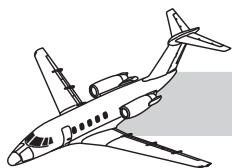
The battery switch must remain in the OFF position during transfer from the left wing to the fuselage tank. Otherwise, the transfer operation will not occur.

WING-TO-WING TRANSFER

A normally closed, solenoid-operated fuel transfer valve is between the left and right electric boost pumps (Figure 5-6). This valve opens when FUEL XFER is selected. To initiate fuel transfer, place the WING FUEL XFER switch in either the LH FROM RH or the RH FROM LH position, and then observe the fuel quantity gauges to verify fuel transfer (Figure 5-11). The transfer rate is approximately 1,400 pounds per hour. To terminate fuel transfer place the WING FUEL XFER switch to the OFF position.

NOTE

If electrical power is lost during transfer, the valve fails in the closed position, and fuel transfer terminates.

**NOTE**

The boost pump in the receiving tank must not be activated or transfer will not take place.

FUSELAGE TANK TRANSFER

To preclude pump cavitation and assure proper fuselage fuel tank transfer, the transfer pumps must be tested prior to 30,000 feet.

To initiate transfer, place the FUS FUEL XFER switch to ON (Figure 5-11). Doing so applies power to the two electric transfer pumps and opens the fuselage tank transfer valve. On some aircraft when the fuselage tank is full, a high-level float switch in the tank illuminates the FUS TANK FULL annunciator. Output from each pump goes into a common manifold and is distributed evenly to both wings.

When fuel transfer is nearly completed, a low-level float switch illuminates the FUS TANK LOW annunciator. As the pumps run dry and the pressure output drops, the FUS TANK XFER FAIL annunciator illuminates. To remove power from the pumps and extinguish all fuselage fuel transfer annunciators, place the FUS FUEL XFER switch to OFF.

CAUTION

The fuselage tank transfer pumps can be damaged if run dry.

Because failure of fuselage fuel to transfer to the wings can compromise airworthiness, redundancy in pumps is required. If the fuselage transfer valve does not open when fuselage transfer is selected, then fuel transfer cannot occur. The valve can be opened manually by pulling the red FUS TANK XFER T-handle under the aft vanity (Figure 5-16). The handle is connected to a cable attached to a lever, which when pulled, opens the valve. This handle cannot be used to manually close the valve.



Figure 5-16. FUS TANK XFER T-Handle

NOTE

SNs 0001—0091 do not have the fuselage tank transfer T-handle.

LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

**EMERGENCY/
ABNORMAL**

For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.



QUESTIONS

1. The incorrect statement concerning the fuel system is:
 - A. The electric fuel boost pump switches do not have to be on for engine start
 - B. With the fuel boost pump switches off, the boost pump activates when the respective starter button is pressed or transfer from that tank is selected
 - C. It is normal for both electric fuel pumps to operate during left-right fuel transfer
 - D. The electric fuel boost pumps automatically activate with the switch in NORM and low pressure sensed in the engine supply line
2. With the FUEL BOOST pump switch in the OFF position:
 - A. The boost pump activates automatically during start and low fuel situations
 - B. The fuel boost pumps activate automatically for start and wing transfer
 - C. The boost pump will not activate under any circumstances
 - D. Both A and B
3. The correct statement concerning the fuel system is:
 - A. With a DC power failure, the primary ejector pump ceases to operate and the engine flames out
 - B. The respective engine should be shut down if the FUEL FLTR BYPASS annunciator illuminates
 - C. The FUEL BOOST pump switches should be on for takeoff and landing
 - D. The primary ejector pump depends on high pressure fuel from the engine-driven fuel pump for operation
4. If the FUEL BOOST ON LH or RH annunciator illuminates, without any action by the crew, and the engine is operating normally, the most probable cause is:
 - A. The low pressure sensing switch has turned the pump on
 - B. The firewall shutoff valve has been closed
 - C. The engine-driven fuel pump has failed
 - D. The FUEL LOW LEVEL switch has turned the pump on
5. To verify that wing fuel transfer is in fact occurring, it is necessary to:
 - A. Monitor the left and right fuel quantity indicators
 - B. Verify that the WING FUEL XFER OPEN light is illuminated
 - C. Ensure that the FUEL BOOST ON LH–RH lights are both illuminated
 - D. Check that the FUS TANK FULL VLV light is not on
6. During transfer of fuselage fuel to the wings:
 - A. The FUS TANK XFER FAIL and the FUS TANK FUEL PUMP 1/2 lights are extinguished
 - B. Illumination of the FUS TANK FUEL PUMP 2 light indicates that the transfer to the right wing has ceased
 - C. The FUS TANK FILL VLV light illuminates
 - D. The FUS TANK FUEL PUMP 1/2 lights illuminate



7. During overwing refueling of the fuselage tank:
- A. Only hot battery bus DC power is required.
 - B. The left FUEL BOOST pump switch must be ON.
 - C. The fuselage tank quantity is monitored by the fuel quantity indicator.
 - D. An external power unit (EPU) must be used.
8. If the wing fuel transfer has been selected and normal DC electrical power is lost:
- A. The system remains in transfer
 - B. The transfer valve fails in the closed position
 - C. The primary ejector pump in the side transferring fuel will replace the electric boost pump
 - D. Both A and C
9. The Citation 650 series aircraft fuel system:
- A. Has low-level float switches in the wing tanks
 - B. Has a low-level float switch in the fuselage tank
 - C. Both A and B
 - D. Requires electrical power for single-point pressure refueling
10. When refueling over the wing:
- A. Fuselage tank fuel is transferred from the right wing tank
 - B. Do not transfer fuel to the fuselage tank while the left wing is being serviced
 - C. The battery switch must be in the BATT position
 - D. External power is required



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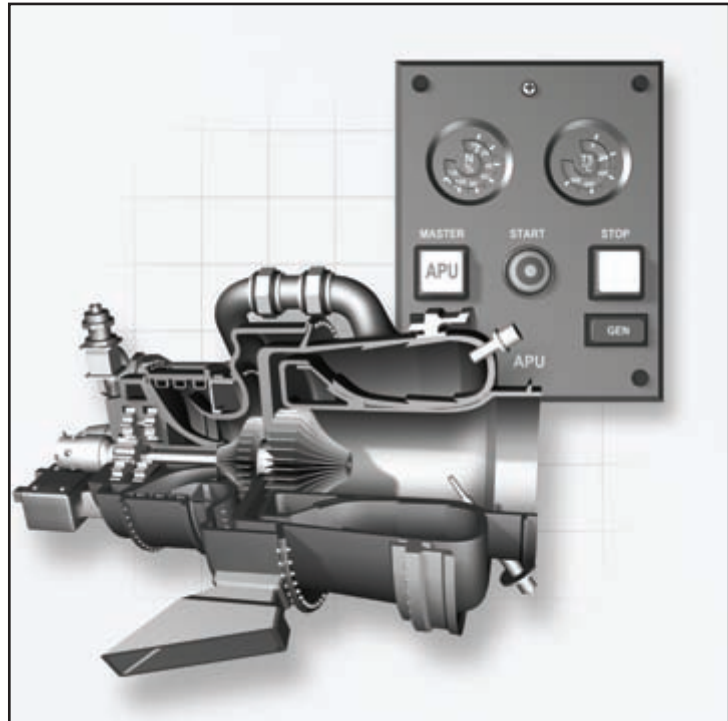
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CHAPTER 6

AUXILIARY POWER SYSTEM



INTRODUCTION

The auxiliary power units (APU) on the Citation 650 series were installed as an optional feature. Various models included ground only or ground/air types. Each APU is a self-contained gas turbine engine in a lightweight stainless steel box in the tailcone compartment, with a portion extending into the baggage compartment.

GENERAL

The APU has self-contained oil, fuel, and ignition systems. During APU startup and operation, an electrical sequencing unit (ESU) monitors the APU, and shuts it down if any primary parameters are exceeded. Separate fire protection, consisting of monitored fire-detection and fire-extinguishing systems, is provided for the APU.

The APU requires normal DC power from the electrical system for starting and requires fuel from the fuel system. Average fuel consumption is approximately 100 to 110 pounds per hour from the right wing.

The APU control panel is either in front of, or behind, the copilot CB panel, depending on



the model. Once started, the APU can provide electrical power, bleed air for air conditioning and service air systems, and hydraulic pressure for all hydraulic systems, if equipped with a hydraulic pump.

For detailed information on the APUs, refer to the FAA-approved AFM and the appropriate supplement for the specific make and model of the aircraft.

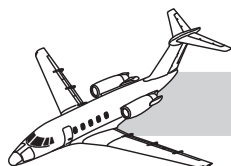
Refer to Table 6-1 for operating limitations for the various APUs.

LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

EMERGENCY/ ABNORMAL

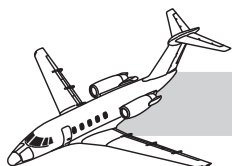
For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.



CITATION 650 SERIES PILOT TRAINING MANUAL

Table 6-1. OPERATING LIMITATIONS

APU	CESSNA GROUND	CESSNA AIR	PATS	PATS
AIRCRAFT	CITATION III	CITATION III	CITATION III	CITATION III
SERIAL NUMBERS	SNs 0001—0104	SNs 0001—0121	SNs 0001—0178	SNs 0001—0178
Operating Limits				
Minimum Fuel Temp For Start	-34°C	-34°C	-54°C to +37°C	-54°C to +37°C
Operating Temperature	-54°C to +37°C	-54°C to +37°C	-54°C to +37°C	-54°C to +37°C
Max Operating Altitude	GROUND ONLY	25,000 FEET	30,000 FEET	30,000 FEET
Thrust Reverser Deployment	30 SECONDS	30 SECONDS	30 SECONDS	30 SECONDS
Maximum Generator Load (Air)	0	200 AMPS	0.5 LOADMETER	0.5 LOADMETER
Maximum Generator Load (Ground)	350 AMPS to ISA -5°C 200 AMPS to ISA +34°C	350 AMPS to ISA -5°C 200 AMPS to ISA +34°C	0.5 LOADMETER	0.5 LOADMETER
Transients (During Engine Start)	>350 AMPS	>350 AMPS	>1.1 LOADMETER	>1.1 LOADMETER
APU Start (Ground)				
Battery Switch A/C Generators On	OFF	ON	OFF	OFF
Battery Switch A/C Generators Off	ON	ON	ON	ON
RH Boost Pump	OFF	OFF	ON	ON
Single Point Refueling	NO	YES	YES	YES
Bus Power Required For Start	RH EXTENSION BUS	RH EXTENSION BUS	HOT BATTERY BUS	HOT BATTERY BUS
Aircraft Generator Assist	NO	YES	NO	NO
Crewmember In Cockpit	NO	NO	YES	YES
Automatic Fire Bottle	YES	YES	NO	NO
Pac Bleed Air SNs 0001—0104	OFF (WITH HYDRAULIC PUMP)	OFF (WITH HYDRAULIC PUMP)	OFF (WITH HYDRAULIC PUMP)	N/A
APU Start (Air)				
Battery Switch	N/A	ON	OFF	OFF
Max Starting Altitude	N/A	25,000 FEET	20,000 FEET	20,000 FEET
Aircraft Generator Assist	NO	NO	NO	NO
Airstart with Batteries	NO	NO	YES	YES
(Within 30 minutes of dual generator failure.)			Dual 44 AMP/HR	Dual 22 AMP/HR (Dual 44 AMP/HR WITH SPZ-8000)
Aircraft Engine Start				
Generator Position for Engine Start	GEN	GEN	OFF	OFF
Other Features				
Hydraulic Pump	YES	YES	YES	NO
APU Generator Voltage	28.5 VOLTS	28.5 VOLTS	28.0 VOLTS	28.0 VOLTS
Bleed Air to PAC Connection	LH/RH PAC VALVES AND ISOLATION VALVE MUST BE OPEN.	LH/RH PAC VALVES AND ISOLATION VALVE MUST BE OPEN.	DIRECT TO LH/RH PAC	DIRECT TO LH/RH PAC
BITE Indicator	YES	YES	YES	YES
Engine Fire Switch Cuts Off APU Fuel	YES	YES	YES	YES
AFM Supplements				
	SUPPLEMENT 7 (NO STC)	SUPPLEMENT 29 (NO STC) SUPPLEMENT 37 (NO STC)	SUPPLEMENT 65 STC SA289NE	SUPPLEMENT 66 STC SA289NE-5


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Table 6-1. OPERATING LIMITATIONS (Cont.)

APU	TURBOMACH	GARRETT
AIRCRAFT	CITATION VI	CITATION VI
SERIAL NUMBERS	SNs 0200 & SUBSEQUENT	SNs 0200 & SUBSEQUENT
Operating Limits		
Minimum Fuel Temp For Start		-34°C
Operating Temperature	-54°C to +37°C	-54°C to +37°C
Max Operating Altitude	30,000 FEET	30,000 FEET
Thrust Reverser Deployment	30 SECONDS	30 SECONDS
Maximum Generator Load (Air)	150 AMPS	150 AMPS
Maximum Generator Load (Ground)	150 AMPS	150 AMPS
Transients (During Engine Start)	>150 AMPS	>150 AMPS
APU Start (Ground)		
Battery Switch A/C Generators On	ON	ON
Battery Switch A/C Generators Off	ON	ON
RH Boost Pump	ON	ON
Single Point Refueling	YES	YES
Bus Power Required For Start	RH EXTENSION BUS	RH EXTENSION BUS
Aircraft Generator Assist	YES	YES
Crewmember In Cockpit	YES	YES
Automatic Fire Bottle	NO	NO
Pac Bleed Air SNs 0001—0104	N/A	N/A
APU Start (Air)		
Battery Switch	ON	ON
Max Starting Altitude	20,000 FEET	20,000 FEET
Aircraft Generator Assist	NO	NO
Airstart with Batteries	YES	YES
(Within 30 minutes of dual generator failure.)	Dual 40 OR 44 AMP/HR	Dual 40 OR 44 AMP/HR
Aircraft Engine Start		
Generator Position for Engine Start	GEN	OFF
Other Features		
Hydraulic Pump	YES	NO
APU Generator Voltage	28.0 VOLTS	28.0 VOLTS
Bleed Air to PAC Connection	DIRECT TO LH/RH PAC	DIRECT TO LH/RH PAC
BITE Indicator	YES	YES
Engine Fire Switch Cuts Off APU Fuel	YES	NO
AFM Supplements		
	SUPPLEMENT 14	SUPPLEMENT 19
	STC SA289NE	STC SA755NE

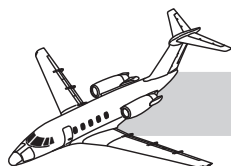


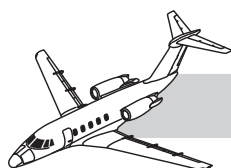
Table 6-1. OPERATING LIMITATIONS (Cont.)

APU	TURBOMACH	GARRETT
AIRCRAFT	CITATION III & VII	CITATION III & VII
SERIAL NUMBERS	SNs 179 & SUBSEQUENT	SNs 179 & SUBSEQUENT
Operating Limits		
Minimum Fuel Temp For Start		-34°C
Operating Temperature	-54°C to +37°C	-54°C to +37°C
Max Operating Altitude	30,000 FEET	30,000 FEET
Thrust Reverser Deployment	30 SECONDS	30 SECONDS
Maximum Generator Load (Air)	150 AMPS	150 AMPS
Maximum Generator Load (Ground)	150 AMPS	150 AMPS
Transients (During Engine Start)	>150 AMPS	>150 AMPS
APU Start (Ground)		
Battery Switch A/C Generators On	ON	ON
Battery Switch A/C Generators Off	ON	ON
RH Boost Pump	ON	ON
Single Point Refueling	YES	YES
Bus Power Required For Start	RH EXTENSION BUS	RH EXTENSION BUS
Aircraft Generator Assist	YES	YES
Crewmember In Cockpit	YES	YES
Automatic Fire Bottle	NO	NO
Pac Bleed Air SNs 0001—0104	N/A	N/A
APU Start (Air)		
Battery Switch	ON	ON
Max Starting Altitude	20,000 FEET	20,000 FEET
Aircraft Generator Assist	NO	NO
Airstart with Batteries	YES	YES
(Within 30 minutes of dual generator failure.)	Dual 40 OR 44 AMP/HR WITH SPZ-8000	Dual 40 OR 44 AMP/HR WITH SPZ-8000 (Dual 22 AMP/HR ON SNs 0082 AND 0152 ONLY)
Aircraft Engine Start		
Generator Position for Engine Start	III – GEN / VII – OFF	OFF
Other Features		
Hydraulic Pump	YES	NO
APU Generator Voltage	28.0 VOLTS	28.0 VOLTS
Bleed Air to PAC Connection	DIRECT TO LH/RH PAC	DIRECT TO LH/RH PAC
BITE Indicator	YES	YES
Engine Fire Switch Cuts Off APU Fuel	YES	NO
AFM Supplements		
	VII – SUPPLEMENT 17 STC SA289NE III – SUPPLEMENT 56 STC SA289NE	VII – SUPPLEMENT 18 STC SA755NE III – SUPPLEMENT 67 STC SA755NE



QUESTIONS

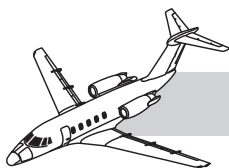
1. Depending on the APU Model, it provides:
 - A. 28-VDC electrical power
 - B. Bleed air for operation of the environmental control units
 - C. Hydraulic pressure to the main system
 - D. All of the above
2. The APU fuel supply is:
 - A. Independent of the aircraft fuel supply
 - B. Drawn from the right fuel tank
 - C. Drawn from the left fuel tank
 - D. Drawn from both fuel tanks
3. A 28.0 VDC APU generator:
 - A. Automatically trips off line when an aircraft generator powers the hot battery bus.
 - B. Cannot be used to assist the batteries during engine starts.
 - C. Is limited to 150 amps maximum under all conditions.
 - D. Can be disconnected from the hot battery bus by pressing the GEN switch-light until it is bright.
4. With the APU operating and the BLEED VALVE switch in OPEN:
 - A. If a PAC O'HEAT light illuminates, no action is required of the crew, as the PAC has automatically shut off the APU bleed air.
 - B. It is not permissible to have the aircraft engine bleed air valves open.
 - C. The door seal inflates whenever the door is closed and locked. If the BLEED VALVE switch is in the CLOSE position, the door seal deflates (aircraft engines not operating).
 - D. Air is supplied to the vacuum ejector for the pressurization control system regardless of the position of the BLEED VALVE switch.



CHAPTER 7 POWERPLANT

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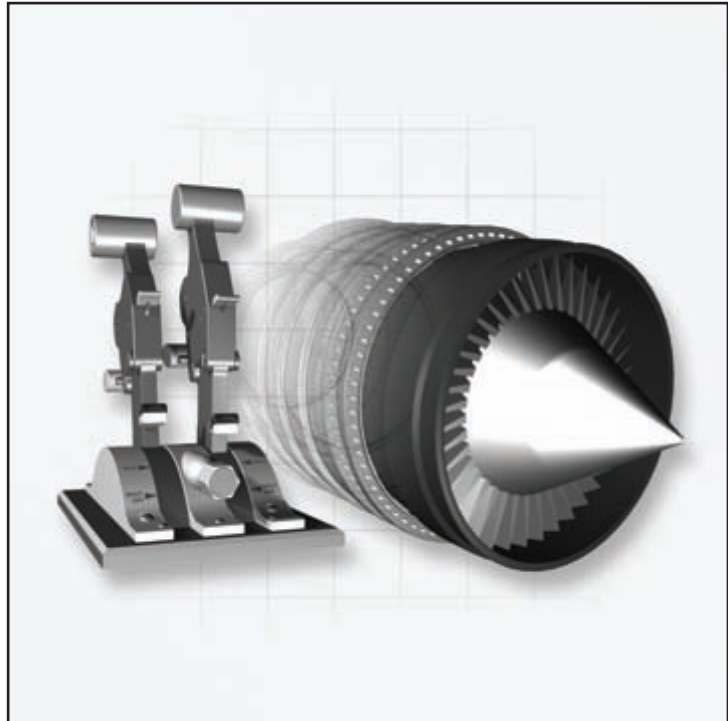


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CHAPTER 7 POWERPLANT



INTRODUCTION

This chapter describes the various powerplants on the Citation 650 series aircraft. Included are descriptions, operation, and limitations of the engine oil, fuel, ignition, instrumentation, and synchronizing systems.

GENERAL

The aircraft are powered by two Garrett TFE731- turbofan engines. The Citation III/VI is equipped with either the TFE 731-3B-100S/3C-100S or TFE731-3BR-100S/3CR-100S. The engines are rated at 3,650 pounds of thrust at sea level (SL). They are interchangeable and can be installed on either pylon. The Citation VII is powered by the TFE731-4R-2S engine, which is rated at 4,080 pounds of thrust at SL.

The TFE731 is a two-spool, medium bypass (3B or 3C) or high bypass (4R) engine with a geared fan, modularized for ease of maintenance. Bypass air cools the turbine section and reduces engine-generated noise, while a reverse-flow annular combustion chamber reduces engine length and weight. Highly efficient fuel scheduling is accomplished by an electro-hydro-mechanical fuel control and an electronic engine computer or a digital elec-



tronic engine computer. Both computers automatically maintain an economical and precise fuel schedule throughout the entire spectrum of atmospheric conditions and thrust requirements. High and low pressure bleed air is extracted from the compressors for the environmental, anti-icing, and service air systems. Figure 7-1 shows a cutaway drawing of a typical TFE-731 engine.

The two shafts (spools) in the engine are concentric, but not connected. The inner low-speed shaft (N_1), driven by three low pressure turbines, turns the four-stage, low pressure axial-flow compressor and the geared fan. The outer high-speed shaft (N_2), driven by the single high pressure turbine, powers the high pressure centrifugal compressor and the accessory gear box.

Air enters the engine through the fan and is immediately divided by a concentric duct (Figure 7-1). Most of the air is directed into the bypass duct, where it passes through stators and is exhausted at the rear of the engine. The bypass ratio is 2.8:1 for 3B and 3C, and 3.1:1 for the 4R. Bypass air accounts for approximately two-thirds of total thrust at sea level, but less than one-half at altitude.

Air entering the inner duct is compressed by the four-stage axial flow compressor and the centrifugal compressor, then passes through the diffuser and into the combustion chamber. The air flows forward, around, and through the annular combustor, where it mixes with fuel from the fuel nozzles. Igniters light the mixture during startup and combustion is continuous. The hot expanding gases reverse direction and drive the high-speed turbine and the three low-speed turbines before passing into the exhaust nozzle.

The amount of static thrust at SL is temperature specific. Below this temperature, rpm is limited to maintain rated thrust. Above this temperature, rpm is limited to preclude engine overtemperature. The pilot must monitor engine rpm and interstage turbine temperature (ITT) to prevent exceeding operating limits.

Engine power control is achieved by throttles in a quadrant on the center pedestal. Throttle

travel is from full aft or CUT OFF, through IDLE to FULL forward, or maximum thrust position. A cutoff stop prevents inadvertent selection of CUT OFF. A latch on the throttle must be raised before the throttle can be moved to or from the CUT OFF position.

A lever to the right of the throttles adjusts throttle friction. Use slow throttle movement (4–6 seconds from idle to takeoff thrust) when conditions permit, to prolong engine life. Rapid power response is available when required. The life of certain engine components is based on engine cycles. Engine cycles are counted in two ways:

- Primary—Start, engine rpm above 80% N_1 , and shutdown
- Alternate—Each landing

If the APR is activated, four engine cycles may be required, depending on the limits reached.

MAJOR SECTIONS

Intake and Fan

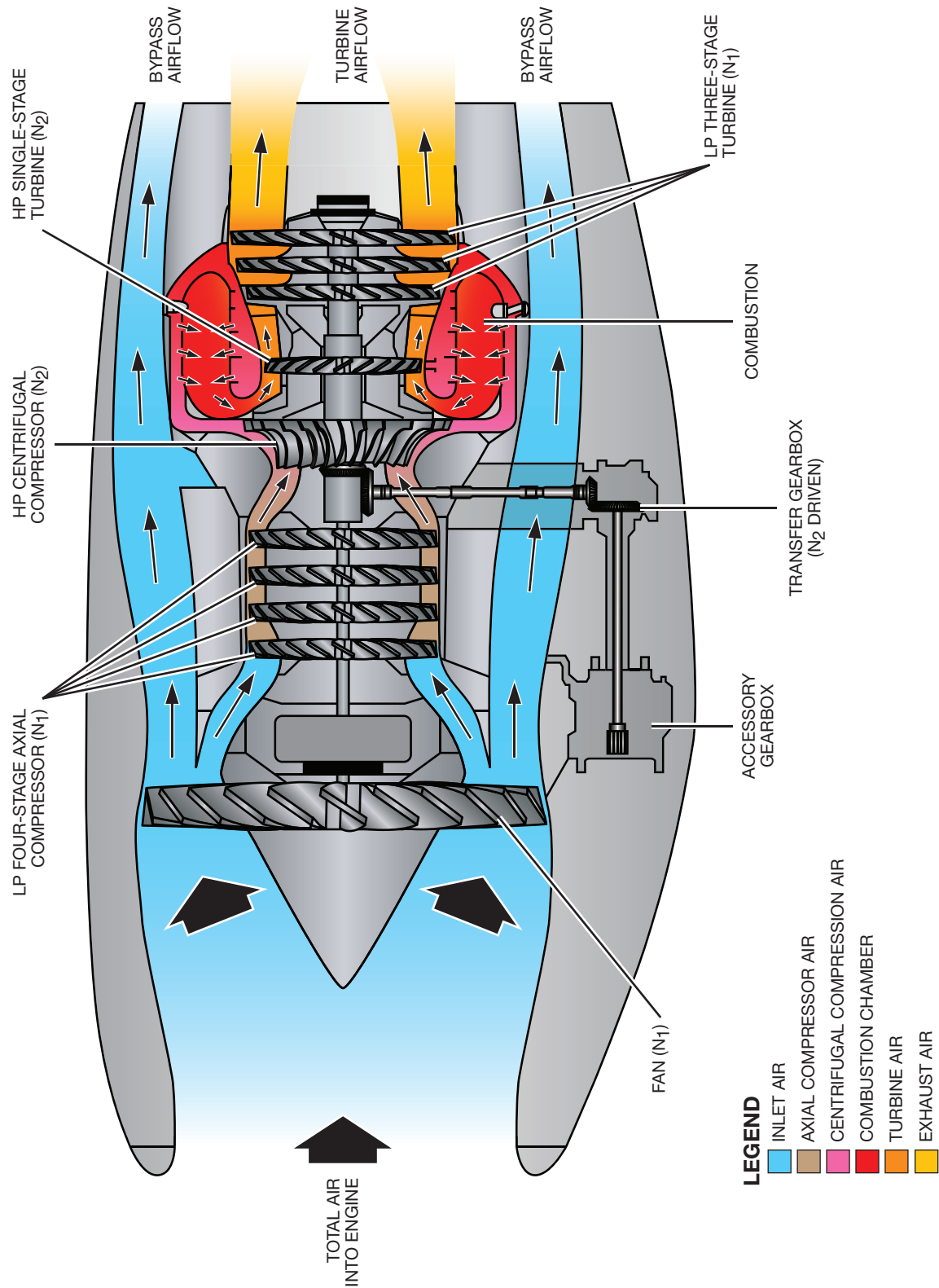
The intake and fan consist of the fan spinner and support, fan rotor assembly and support, planetary gear assembly, low-pressure compressor stator, bypass inlet housing, and bypass stator. The inlet housing is noise attenuating and armored around the fan for blade containment. The low-speed shaft drives the fan through planetary reduction gears, which limit fan N_1 to 101.5%. The fan spinner is non-icing by design. Fan rotation is opposite all other rotating components in the engine. The outer diameter of the fan accelerates a moderately large air mass at a relatively low velocity into the full length bypass duct. The inner diameter accelerates the air mass into the four-stage axial-flow compressor.

Compressor Assembly

The compressor assembly consists of a low pressure (LP) compressor and a high pressure (HP) compressor. The LP compressor is a four-stage (alternating with stators) axial compressor, driven by the inner low-speed turbine shaft (N_1). The LP compressor increases the



7 POWERPLANT





air mass pressure and accelerates it rearward. The HP compressor is a single-stage centrifugal compressor driven by the outer high-speed turbine shaft (N₂). It further increases the pressure of the air mass and directs it rearward into the combustor.

Surge Bleed Valve

A tulip-shaped surge bleed valve on the right side of the engine is opened pneumatically by the fuel computer under surge conditions to increase the surge margin. The open valve vents air from the front face of the impeller into the bypass duct. The valve opens one-third when the fuel computer is not operating. The valve opens completely when the throttle is at IDLE and closes just above IDLE when the fuel computer is operating.

During engine acceleration, when combustor pressures are high, or if the HP and LP compressor speeds mismatch, the low pressure compressor can stall or surge because of excess air on the face of the impeller.

Combustor

The combustor consists of a reverse-flow annular combustion chamber aft of the HP compressor and surrounding the engine turbines. The combustion chamber controls mixing of fuel and air, provides containment for the combustion gases, and directs them for expansion through the turbine. The combustion chamber includes 12 equally spaced duplex fuel nozzles and two high-energy igniter plugs.

Turbine

The turbine section consists of a single high pressure turbine wheel and three low pressure turbine wheels. The high pressure turbine wheel powers the high-pressure compressor and accessory drive. The low pressure turbine wheels power the axial compressor and fan. Nozzles or stators are ahead of each wheel.

Ten temperature probes, connected through a harness between the high pressure turbine and low pressure turbine, provide ITT indications for the fuel computer and cockpit indicator.

Gases leaving the turbine section exhaust to the atmosphere through the exhaust nozzle.

Exhaust

The exhaust section consists of the primary and bypass air exhaust ducts. The primary exhaust section directs the combustion gases to the atmosphere. The bypass air exhaust directs the fan bypass air to the atmosphere.

Accessory Drive Gearbox

The accessory drive gearbox drives the starter-generator, engine-driven fuel pump, hydraulic pump, fuel control, oil pump, and AC alternator. A vertical tower shaft, driven by the high-speed (N₂) shaft, extends down to the transfer gearbox near the middle of the engine. A horizontal shaft extends forward from the transfer gearbox to the accessory drive gearbox under the front of the engine. Figure 7-1 shows the major sections of the engines.

COMPONENTS

Oil System

The lubricating system consists of an oil pump on the accessory gear box and an oil reservoir on the engine. The lubricating system provides cooled, clean oil at nearly constant pressure to the engine bearings, planetary gearcase gears and bearings, transfer case, and accessory drive.

Oil Reservoir

The oil reservoir contains 6 quarts of oil, but overall engine capacity is 11.6 quarts. Use only approved brands of oil listed in the *AFM*.

Oil Pump

The oil pump consists of a pressure pump, four scavenge pumps, a pressure regulator, and a chip detector.

Oil is drawn from the oil tank by the pressure pump. Pump output and pressure is maintained above 68% N₂ rpm by a pressure regulating valve. Oil is routed through a micron filter. If the filter becomes blocked, a bypass valve opens to ensure engine lubrication (Figure 7-2).



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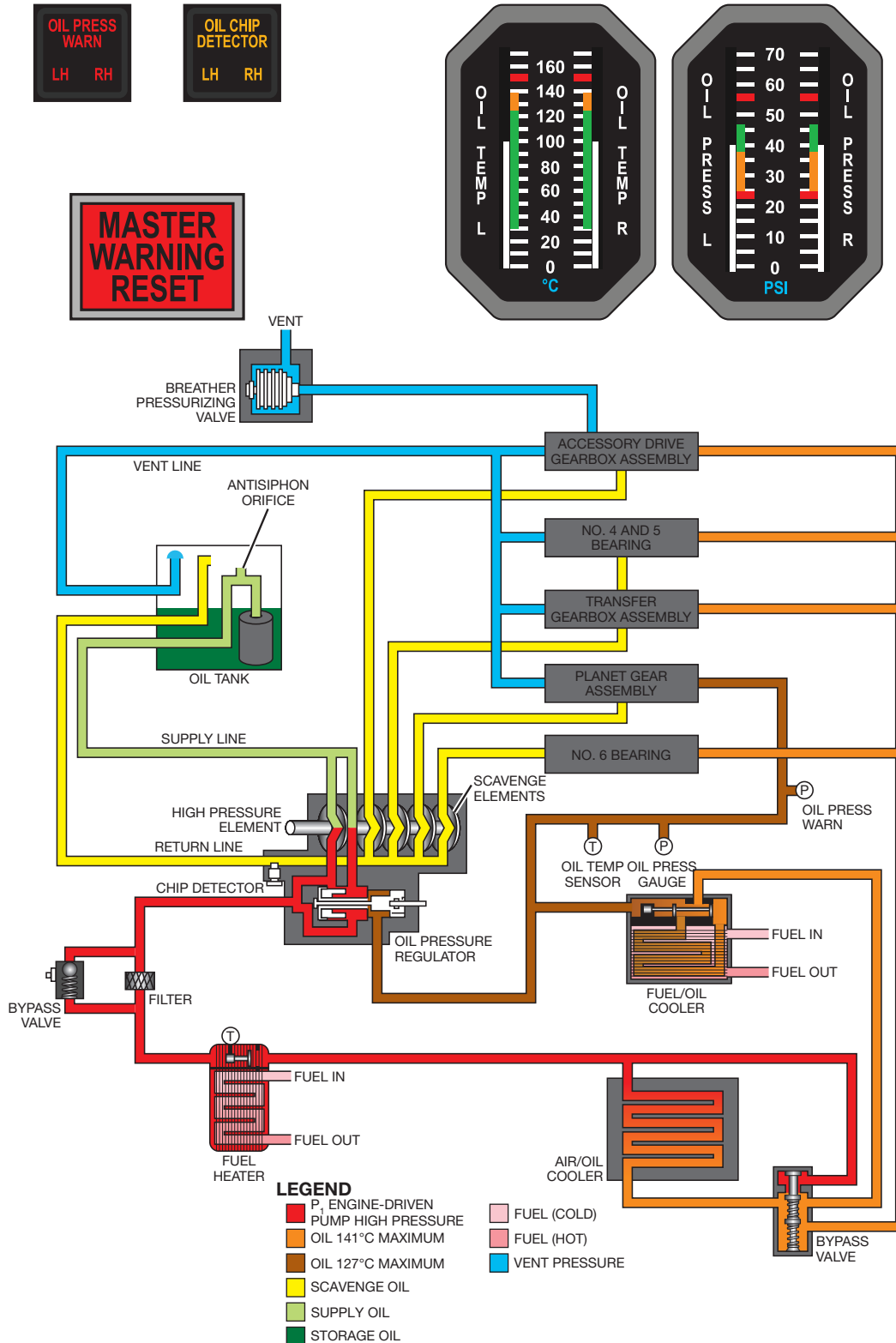
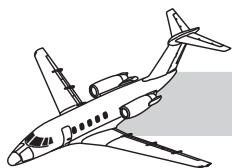


Figure 7-2. Oil System



The oil is then routed to an oil/fuel heater and then to a temperature control and bypass valve. The valve modulates with temperature changes until it closes completely, forcing the oil through three, segmented air/oil coolers in the engine bypass duct. If oil flow is blocked in the air/oil coolers, the valve opens, allowing oil to bypass the coolers.

Leaving the air/oil coolers, the oil is directed to the transfer and accessory gear boxes, and the cavities for No. 4, No. 5, and No. 6 bearings. Oil to the planetary gearcase is routed through a fuel/oil cooler and lubricates bearings 1, 2, and 3. Oil leaving the air/oil coolers is maintained at a maximum of 141°C (285°F). The temperature is further reduced in the fuel/oil cooler to a maximum of 127°C (260°F). Oil temperature is sensed downstream of the fuel/oil cooler and is displayed on a dual vertical tape gauge.

LH-RH OIL PRESS WARN



Oil pressure is sensed downstream of the fuel/oil cooler. A pressure switch senses the pressure and illuminates the OIL PRESS WARN LH–RH annunciator if oil pressure is lower than 25 psi, as do the MASTER WARNING RESET switchlights. A transducer also senses oil pressure and displays it on a dual vertical tape gauge. The tape and light are independent indications and need not agree.

LH-RH OIL CHIP DETECTOR



The four scavenge elements of the oil pump return oil to the oil tank. The OIL CHIP DETECTOR illuminates if metal chips are detected by the metal chip detector in the line between the scavenge elements and the oil tank.

The oil tank, planetary gearcase, No. 4 and No. 5 bearing cavities, and transfer gearbox all vent to the accessory gear box. The accessory gear box vents to the atmosphere through the breather pressurization valve, which opens at low altitude but closes between 27,000 to 30,000 feet for higher altitude operations.

Operation

Engine oil level must be checked within one hour of engine shutdown to obtain an accurate oil level.

To check oil quantity, check the sight gauge on the right side of each engine (Figure 7-3). Add oil to the right engine via a cap on the oil tank next to the sight gauge. Add oil to the left engine via the oil filler tube on the left side of the engine. Check the oil filter bypass indicator during preflight and postflight inspections (Figure 7-4).



Figure 7-3. Oil Servicing

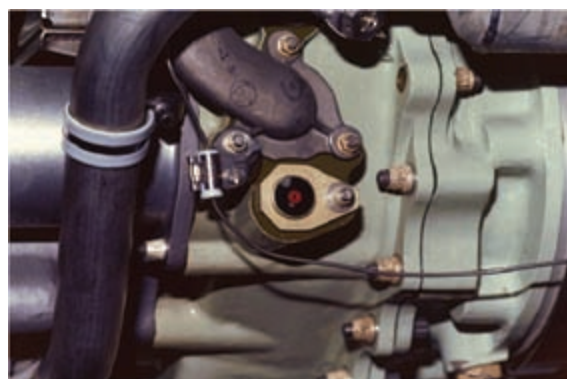


Figure 7-4. Oil Filter Bypass Indicator



Fuel System

The engine fuel system consists of the:

- Engine-driven fuel pump
- Fuel heater
- Electro-hydro-mechanical fuel control
- Electronic engine computer (EEC) or digital electronic engine computer (DEEC)
- Fuel flow transmitter
- Fuel flow divider
- Manifolds
- Nozzles
- Fuel/oil cooler (covered in Oil System).

Engine operation is controlled by the fuel control unit (FCU), which contains fuel shutoff and metering sections. This unit, which is on the fuel pump, provides the throttle linkage connection point and N₂ overspeed protection when the fuel computer is in use.

During normal operation, the fuel computer in the tailcone sets the thrust, governs speed, and limits acceleration and deceleration through electrical inputs to a metering valve in the FCU. If the DC electrical bus or the computer fails, or if the pilot chooses, the FCU assumes control of the engine.

Engine-Driven Fuel Pump

The fuel pump provides high pressure to the FCU and motive flow fuel for the primary ejector pump in the fuel system. The fuel pump houses:

- Centrifugal low pressure boost pump
- Filter element
- Filter bypass valve with electric signal to the cockpit
- Anti-ice valve
- Vane-type high pressure pump
- High pressure relief valve
- Motive-flow lockout valve

- Motive-flow pressure regulator

The high pressure pump has a greater capacity than is needed to operate the engine under all conditions. For this reason, pump discharge pressure is limited by the high pressure relief valve. The motive flow lockout valve blocks flow to the motive flow pressure regulator until fuel is furnished to the FCU. When this valve opens, the motive flow pressure regulator limits motive flow pressure to 300 psi. Motive flow is covered in Chapter 5—"Fuel System."

Oil/Fuel Heater

Fuel leaving the fuel filter in the pump passes over a temperature sensitive element of the anti-ice valve. If the fuel temperature is below set limits, the valve modulates open and directs some fuel returning from the fuel control through the oil/fuel heater until the valve fully opens. The fuel returns to the inlet of the fuel filter to keep it free of ice.

If fuel inlet temperature exceeds set limits, all return fuel goes directly to the high pressure pump inlet. The oil/fuel heater is a heat exchanger through which hot engine oil passes continuously. When the anti-ice valve opens, return fuel passes through the coils, transferring heat from the oil to the fuel.

Fuel Control Unit

The FCU contains:

- Fuel metering section
- Torque motor and valve (for computer fuel metering)
- Ultimate overspeed solenoid
- Throttle angle potentiometer
- Shutoff valve
- Compressor discharge limiter
- N₂ overspeed governor

The governor limits turbine rpm to 105% and acts as an onspeed governor in manual operation. In normal operation, the metering section of the fuel control is fully open, and fuel



metering is controlled by the computer through the torque motor. The shutoff valve in the fuel control stops fuel flow when the engine is shut down with the throttle.

Flow Divider, Fuel Manifolds, and Nozzles

The flow divider controls fuel flow to the primary or secondary nozzles. The primary nozzles operate continuously; the secondary nozzles operate when additional fuel is needed at higher power settings.

ELECTRONIC ENGINE COMPUTER (CITATION III/VI ONLY)

Description

The EEC is in the tailcone maintenance compartment (Figure 7-5). The EEC contains schedules for starting, idle, acceleration, take-off power, climb/cruise, and deceleration as well as surge protection, which controls the surge bleed valve. The EEC schedules fuel flow from the FCU by accounting for throttle angle, fuel density, and atmospheric conditions and provides both rpm governing and temperature limiting protection.



Figure 7-5. Engine Fuel Computer

The EEC requires between 14–30 VDC, controlled through the LH–RH FUEL COMP MAN/NORM cockpit switch on the tilt panel. Inputs to the fuel computer are engine inlet

temperature (T_{T2}), pressure (P_{T2}), ITT, N_2 , and N_1 rotor speeds, and throttle angle. Based on these inputs the EEC furnishes an appropriate electrical signal to the torque motor in the FCU, which regulates fuel flow to the engine.

The EEC operates satisfactorily with voltage as low as 14 volts. If voltage decreases below 14 volts, the EEC automatically transfers to manual mode. Manual mode also may be selected by placing the control switch to MAN.

Within the EEC, two idle rpm schedules control minimum rpm when the throttles are at idle. High idle rpm is selected automatically when the aircraft is in flight (squat switch logic). Lower idle rpm is selected on the ground if the ground idle switch is in NORM. Specific idle rpm is not one distinct value but varies with changes in altitude and air temperature. Idle rpm increases as outside air temperature (OAT) or altitude increases.

The EEC automatically adjusts fan speed to maintain a flat-rated thrust. If the ambient pressure/temperature relationships cause the ITT to exceed limits, then fan speed is decreased and rated thrust is not available.

The EEC has a “flat spot” near idle, which is the first few degrees of throttle movement above IDLE that produce little or no change in engine speed. At high engine speeds, speed is proportional to throttle position.

Each computer is controlled by a MAN/NORM switch in the cockpit. The FUEL COMP MANUAL annunciator indicates computer failure or that the switch is in the MAN position.

If the annunciator illuminates or if an electrical failure occurs, the automatic functions of the EEC are not available and the engine operates in manual mode. The computer is self-monitoring and generally reverts to manual mode if it malfunctions.

When the EEC is in manual mode, the surge bleed valve opens one-third to prevent compressor stalls and the flyweights in the FCU are used as an onspeed governor in conjunction



with throttle position. Acceleration is slow and throttle misalignment is expected. As such, the crew must monitor all engine limits and make adjustments as necessary.

NOTE

Fuel computers are required for normal operation. To fly the aircraft with an inoperative computer, a ferry permit must be obtained. The mode switch on the fuel computer must be positioned to MANUAL and the cockpit switch positioned to NORM in order to furnish normal ultimate overspeed protection for N_1 and N_2 .

Fuel Computer Operation

Fuel Computer Switch—NORM

NORM controls the engine through all phases of flight. The mechanical N_2 governor in the FCU is at 105% to provide overspeed protection.

Start

Start schedule is controlled by N_2 rpm. As rpm increases, fuel flow increases to allow engine acceleration, but overfueling is prevented to avoid overtemperature. Enrichment is part of the schedule to increase fuel pressure and, therefore, fuel flow at low rpm. Enrichment, which is automatic on start to 200°C (ITT), improves atomization at light-off and acceleration at low temperatures. ITT is in the start schedule for temperature limiting. If ITT is high, fuel flow is reduced to keep the temperature within limits.

Governing Schedule

Governing is controlled by N_2 rpm and throttle angle inputs. The governor senses throttle angle by a potentiometer in the FCU and drives the torque motor (fuel flow) to maintain N_2 rpm. ITT provides an input for temperature limiting. Idle speed is adjusted as a function of inlet temperature and pressure. Speed is limited by flat-rated thrust or turbine temperature.

The acceleration schedule controls N_1 rpm. During slow to moderate throttle increases the acceleration schedule provides more fuel to increase N_1 rpm.

During fast throttle advances, the surge schedule acts as a limiter to protect the engine from surge. The computer senses N_1 , N_2 , and torque motor current (fuel flow). If the torque motor current gets too high for N_1 or N_2 , the surge schedule opens the surge valve. The surge schedule also opens the surge valve whenever N_2 rpm becomes too low in relation to N_1 rpm. The surge bleed valve is open at idle.

Overspeed protection provides temporary fuel shutoff in the FCU (ultimate overspeed solenoid).

The minimum fuel schedule provides a deceleration schedule to prevent a blowout or a serious mismatch between the speeds of the two spools.

The EEC monitors all computations, inputs, and outputs (except ITT) and automatically switches to manual if the inputs and outputs are out of tolerance.

DIGITAL ELECTRONIC ENGINE COMPUTER

Description

The DEEC (Figure 7-6) is in the tailcone maintenance compartment. The DEEC contains schedules for starting, idle, acceleration, take-off power, climb/cruise, and deceleration as well as surge protection, which controls the surge bleed valve. The DEEC schedules fuel flow from the FCU by taking into account throttle angle, fuel density, and atmospheric conditions, and provides both rpm governing and temperature limiting protection.

The DEEC requires 13–30 VDC, controlled through the FUEL COMP MAN/OFF/ NORM switch on the cockpit tilt panel. Inputs to the DEEC are engine inlet temperature (T_{T2}), pressure (P_{T2}), ITT, N_2 and N_1 rotor speeds, and throttle angle. Based on these inputs the DEEC furnishes an appropriate electrical signal to



Figure 7-6. Digital Electronic Engine Computer

the torque motor in the FCU, which regulates fuel flow to the engine.

The modern digital circuitry in the DEEC uses fewer components and has increased reliability compared to earlier analog engine computers.

The DEEC operates satisfactorily with voltage as low as 13 volts. If voltage decreases below 13 volts, the DEEC automatically transfers to manual mode. Manual mode also may be selected by placing the control switch to MAN.

Within the DEEC, two idle rpm schedules control minimum rpm when the throttles are at idle. High idle rpm is selected automatically when the aircraft is in flight (squat switch logic). Lower idle rpm is selected on the ground if the ground idle switch is in NORM.

The DEEC calculates an N_2 rpm for a specified ambient air temperature and altitude, based on two schedules: a flat rated schedule and a maximum speed schedule. However, N_1 is restricted to a maximum value of 101.5% by the N_1 limiter in the DEEC.

At higher air temperature and altitude, maximum thrust cannot be achieved prior to reaching the maximum permissible ITT or N_2 rpm. Maximum thrust can be obtained when operating on the flat-rated schedule. When the aircraft is operating on the flat-rated schedule, as altitude increases, N_1 also increases to offset the lower density and provide the same flat-rated thrust.

At altitudes greater than 10,000 feet, the maximum speed schedule is lower than the flat rated schedule and acts as the controlling schedule. At higher ambient air temperatures, ITT is the limiting factor. Therefore, rpm is decreased so that maximum ITT is not exceeded. In cases of both extreme altitude and high air temperatures, maximum rated thrust is not achieved because of rpm and ITT limits.

The DEEC incorporates climb and cruise schedules. Setting the throttle for climb power early in the climb normally does not require periodic throttle adjustments during climb. The DEEC maintains a scheduled N_2 designed to give N_1 values close to the N_1 rpm specified in the *AFM* climb power charts. The DEEC uses a similar schedule for cruise power settings.

Fuel Computer Operation

Fuel Computer Switch—NORM

NORM controls the engine through all phases of flight. The mechanical N_2 governor in the FCU is at 105% to provide overspeed protection.

Start

The start schedule is controlled by N_2 rpm. As rpm increases, fuel flow increases to allow engine acceleration, but overfueling is prevented to avoid overtemperature. Enrichment, which is automatic on start to 200°C (392°F) (ITT), improves atomization at light-off and acceleration at low temperatures. Enrichment is part of the schedule for increasing fuel pressure and fuel flow at low rpm. ITT is in the start schedule for temperature limiting. If the ITT is high, fuel flow is reduced to keep the temperature within limits.

Governing Schedule

Governing is controlled by N_2 rpm and throttle angle inputs. The governor, which senses throttle angle by a potentiometer in the FCU, drives the torque motor (fuel flow) to maintain N_2 rpm. ITT provides an input for temperature limiting. Idle speed is adjusted for inlet temperature and pressure. Speed is limited by flat-rated thrust or turbine temperature.



The acceleration schedule controls N_1 rpm. During slow to moderate throttle increases the acceleration schedule provides more fuel to increase N_1 rpm.

During fast throttle advances, the surge schedule acts as a limiter to protect the engine from surge. The computer senses N_1 , N_2 , and torque motor current (fuel flow). If the torque motor current becomes too high for N_1 or N_2 , the surge schedule opens the surge bleed valve. The surge schedule also opens the surge bleed valve if N_2 rpm becomes too low in relation to N_1 rpm. The surge bleed valve is open at idle.

The DEEC incorporates an ITT limiter feature, which reduces fuel flow if the ITT exceeds limits. The feature has limited authority and primarily prevents excessive ITT overshoot during rapid throttle advancement.

The engine-mounted FCU has an ultimate overspeed solenoid, controlled by the DEEC in either NORM or MAN mode. The solenoid, when activated, closes the shutoff valve in the FCU, regardless of throttle position. The ultimate overspeed solenoid energizes if N_1 exceeds limits. The automatic shutdown feature is designed to protect the airframe if a catastrophic engine failure occurs.

NOTE

The DEEC does not automatically limit N_1 for the TFE731-4-2S engines.

The FUEL COMP MANUAL LH–RH annunciator illuminates if the FUEL COMP switch is placed to MAN or if the DEEC fault detection logic automatically transfers the DEEC to manual mode. When the DEEC is in manual mode, the manual mode solenoid is deenergized, which allows the throttle position to set or modulate the flyweight governor in the FCU. In manual mode, the throttle position thus determines the onspeed governor. Power response to throttle movement is considerably slower in manual mode. Care must be taken to ensure that engine limits are not exceeded.

The surge bleed valve is deenergized to one-third open, allowing conservative operation in manual mode for compressor stall margin.

The ultimate overspeed solenoid is still controlled by the DEEC when the DEEC is in manual mode.

NOTE

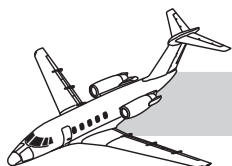
The DEECs are required for normal operation. To fly the aircraft with an inoperative computer a ferry permit must be obtained. The cockpit fuel computer switch must be positioned to MANUAL to provide control of the ultimate overspeed solenoid for N_1 and N_2 overspeed protection.

Positioning the FUEL COMPUTER switch to OFF removes all power from the DEEC and illuminates the FUEL COMP MANUAL LH–RH annunciator. The OFF position is primarily for maintenance functions only.

ENGINE SYNCHRONIZATION

The engine synchronizer compares engine speeds to generate a trim signal to the electronic fuel computer of the slave (right) engine. As a result, the fan or turbine speed for the slave engine is synchronized with the master (left) engine. The synchronizer control is a three-position rotary switch on the pedestal. Selections are FAN, OFF, and TURB (Figure 7-7). An indicator above the switch shows right engine speed relative to the left engine when the system is in use. The synchronizer system will not function if one computer is in the manual mode, but the speed difference will be displayed.

When the synchronizer switch is moved from the OFF position, a small green light near the switch illuminates. In addition, because use of the synchronizer for takeoff or landing is prohibited, an amber ENG SYNC light under the N_2 display illuminates if the synchronizer is engaged and the FLAP handle is out of the UP position.



AUTOMATIC PERFORMANCE RESERVE

The automatic performance reserve system (APR) provides engine reserve power if either engine loses thrust during takeoff. APR engines are designated TFE731-4R, TFE731-3BR/3CR.

The APR system works in conjunction with the engine fuel computers (EEC or DEEC) and the engine speed synchronizer unit. The computers interact with the APR/synchronizer systems for control.

APR ON Switchlight

The APR ON switchlight, when pressed, activates the APR system, and illuminates to indicate the system is in operation (Figure 7-9). If armed, it also illuminates when the APR system senses 5% turbine speed (N_2) differential and APR thrust takes over, if thrust is set to the proper takeoff thrust setting prior to thrust loss.

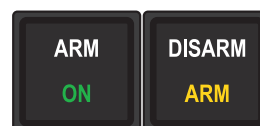


Figure 7-9. APR ON and ARM/DISARM Switchlights

ARM/DISARM Switchlight

The ARM/DISARM switchlight activates (arms) or deactivates (disarms) the APR system (Figure 7-9). The switchlight illuminates when the system is armed.

To arm the APR system push the APR ARM/DISARM switchlight, and then verify that the APR ARM annunciator illuminates.

To manually activate the APR system, press the APR ON switchlight, and then verify that the APR ON annunciator illuminates.

To check the system during taxi, verify that both throttles are at IDLE, press the APR ARM/DISARM switch, and then verify that the APR ARM annunciator illuminates. Advance

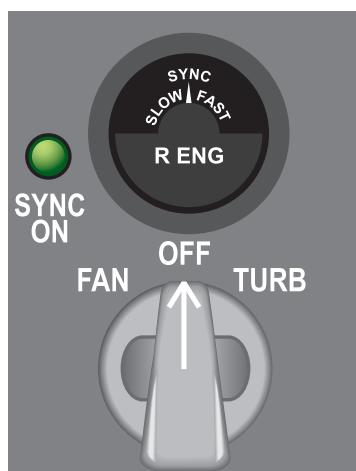


Figure 7-7. Synchronizer Switch and Indicator

GROUND IDLE

A GND IDLE switch has NORM and HIGH positions (Figure 7-8). In NORM, engine idle rpm is reduced approximately 4% N_2 during ground operation. During landing, this is delayed 8 seconds after touchdown. Ground idle reduces braking required for taxiing. A GROUND IDLE annunciator light illuminates when the lower idle limit is in effect. This is a malfunction in flight which can be corrected by positioning the switch to HIGH. The HIGH position must be used for touch-and-go landings as per the *AFM*.

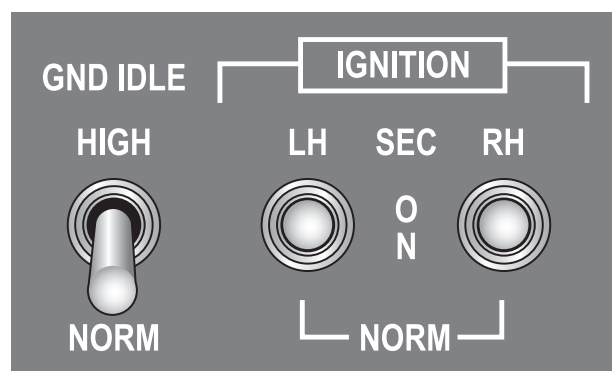


Figure 7-8. GND IDLE and IGNITION Switches



one throttle to 5% to 10% above IDLE, and then verify that the APR ON light illuminates.

Reduce both throttles to IDLE, and then note the resulting N_2 speed. When the APR ARM/DISARM switch is pressed, both N_2 speeds normally decrease approximately 1% and the APR ARM and APR ON lights extinguish.

Before takeoff, verify that the APR ARM annunciator is illuminated, indicating that the APR system is armed.

To disarm the system after takeoff, press the APR ARM–DISARM switchlight, and then verify that the APR ARM annunciator is extinguished.

IGNITION SYSTEM

The ignition system converts DC electrical input to high-voltage DC output, providing electrical sparks necessary for engine ignition.

The ignition system consists of an exciter box, two ignition leads, and two spark igniters, which operate from a 10–30 VDC supply.

The exciter is at the 10 o'clock position on the engine (Figure 7-10). It converts DC electrical input to high-voltage DC output through a solid-state capacitance discharge circuit. If one igniter fails, the other is adequate for normal ignition.



Figure 7-10. Ignition Exciter

Ignition Switches

The LH–RH IGNITION switches on the center switch panel control power to the exciter box (Figure 7-8).

When the switch is in the SEC or ON position, the exciter box receives power continuously, and ignition is continuous. Power for the SEC position and engine anti-ice is from the main feed buses in the baggage compartment. Power for the ON position is from the left extension and right branch buses on the left CB panel in the cockpit. Starting ignition comes from the hot battery bus in the baggage compartment.

The NORM position furnishes ignition automatically for engine start and when engine anti-ice is selected. During engine start, ignition occurs after the start button is pressed and the throttle is brought to IDLE. Ignition is terminated at 42% to 48% N_2 rpm by the speed-sensing switch on the starter-generator.

Set the ignition switches to ON for takeoff, landing, practice stalls, flying through heavy rain or turbulence, and emergency descents.

L–R IGN Annunciator

The L–R IGN, a small green annunciator near the top of the ITT instrument panel, illuminates when the exciter is receiving electrical power (Figure 7-11).

INSTRUMENTATION

Powerplant instrumentation is on the top center instrument panel and includes dual vertical tapes and LED displays (Figure 7-11).

N_1 (Fan Speed)

The N_1 monopole pickup measures the rotation speed of the low-speed shaft. The monopole acts as an AC electrical generator, generating voltage as the gear teeth move through its magnetic field. One output is displayed in the cockpit indicator, which is powered by the emergency branch bus; another output is furnished to the engine fuel computer (Figure 7-11).

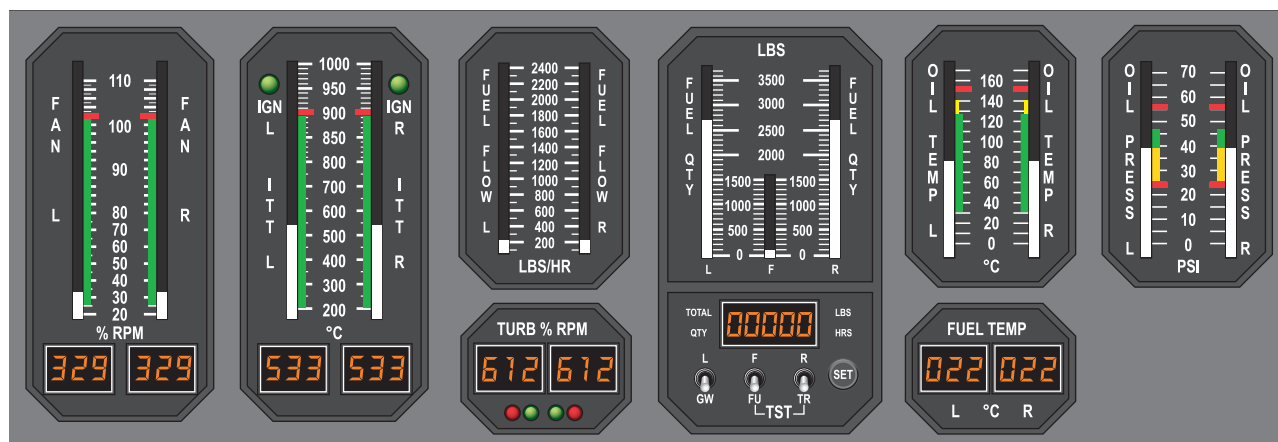
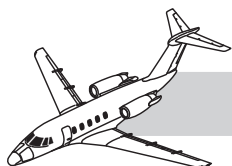


Figure 7-11. Engine Instrument Panel

Interstage Turbine Temperature (ITT) (T_{T5})

Ten thermocouples between the high and low pressure turbine sections provide ITT outputs for the fuel computer and cockpit indicator. (Figure 7-11). The ITT instruments use 28 VDC power from the emergency branch bus.

N_2 (Turbine Speed)

The N_2 monopole pickup is similar to the N_1 system but is on the bottom of the transfer gearbox, which is driven by the high-speed shaft (Figure 7-11). One output is for cockpit display and another output is for the fuel computer.

Fuel Flow

The fuel flow indicating system consists of a fuel flow transmitter on each engine and a dual vertical tape indicator on the instrument panel (Figure 7-11).

OPERATION

STARTING

Normal Start Procedure

Pressing the start button engages the start relay, activates the fuel boost pump, arms the ignition, and illuminates the cockpit floodlights.

At 10% N_2 and N_1 indication, move the throttle to IDLE; this provides the engine fuel and ignition. The ignition light illuminates. The ITT increases within 10 seconds, indicating light-off, followed within 10 seconds by the oil pressure indication.

At 20% N_2 , N_1 increases. The start automatically terminates at 42% to 48% N_2 , followed by the generator coming on line, if the generator switch is in GEN. Engine speed increases to 56% N_2 at idle with the GND IDLE switch in NORM.

Reasons to terminate the start include:

- No N_2 rpm
- N_2 rpm less than 10% in 6 seconds
- No ITT 10 seconds after throttle to IDLE
- No N_1 rpm at 20% N_2
- N_1 or N_2 rpm not increasing to idle
- ITT of 890°C for 3B; 910°C for 3C; 952°C for 4R
- No oil pressure within 10 seconds after light-off
- Not at idle within 50 seconds after light-off
- Unusual noise or vibration
- Uncontrollable acceleration beyond idle



If the start is terminated, place the throttle in CUT OFF and motor the engine for at least 15 seconds to clear any fuel in the combustion chamber.

Cold Start

In extremely cold conditions, N_1 rpm may not register at 20% N_2 , even if the fan is rotating. Observe the fan during start to verify rotation. N_1 indications are normal at idle.

Restart on Ground

If engine restarts are required within 20 to 45 minutes after engine shutdown, then 10 minutes after shutdown either rotate the fan by hand several times or motor the fan for 5 seconds using the starter.

Air Start

An air start may be windmilling or starter-assisted. The starter must be used when stabilized N_2 rpm is below 15% (Figure 7-12).

MANUAL FUEL CONTROL GOVERNOR CHECK

Check the manual fuel control governor prior to taxi with brakes applied. If performing the check during taxi, delay the check until clear of any obstructions or personnel. Devote full attention to the check, and perform the check according to the FAA-approved *AFM*.

To check the manual fuel control governor, set one FUEL COMP switch to MAN, and then check for proper changeover to manual mode. Advance and retard the throttle and verify that N_2 rpm follows throttle movement. Set the FUEL COMP switch to NORM, and then repeat the procedure for the other engine.

If any abnormality is noticed, such as a rapid increase in engine response to throttle movement, set the FUEL COMP switch to NORM immediately. Leave one hand in control of the computer switch and leave the other hand on the throttle in case the throttle must be placed in CUT OFF to prevent an overspeed condition.

SHUTDOWN ON THE GROUND

Allow the engine to idle (below 38% N_1) for two minutes prior to shutdown. Coast-down time is about 70 seconds for N_1 and 40 seconds for N_2 .

PERFORMANCE TRENDS

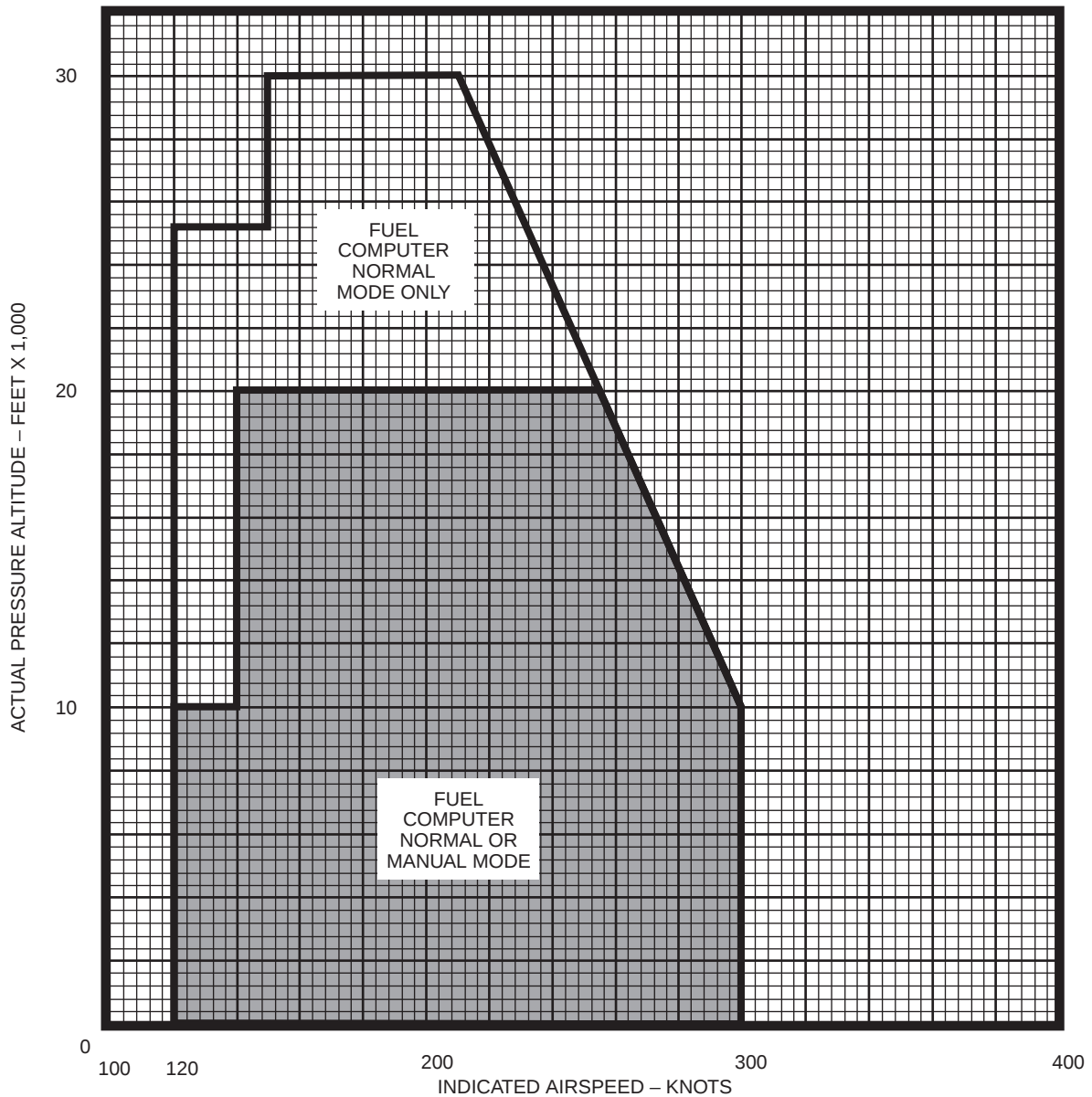
It is recommended that engine parameters be recorded for noting changes that may be occurring in the engine. A flight check form is available from the Garrett Turbine Engine Company for this purpose.

LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

EMERGENCY/ABNORMAL

For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.



NOTES:

1. ENGINE WINDMILLING AIRSTART REQUIRES A STABILIZED MINIMUM TURBINE SPEED OF 15% TURBINE RPM (N_2).
2. IF TURBINE SPEED IS NOT STABILIZED OR IF AIRSPEED RESULTS IN A TURBINE SPEED LESS THAN 15%, THEN STARTER ASSIST IS RECOMMENDED.
3. MAXIMUM ALTITUDE FOR MANUAL MODE AIRSTARTS IS 20,000 FEET.

Figure 7-12. Airstart Envelope

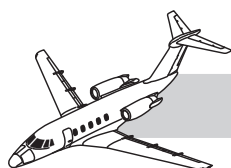


QUESTIONS

1. Most of the thrust produced by the TFE731 engine is obtained from:
 - A. Expanding gases from the combustion chamber only
 - B. The fan, which is driven by the high pressure turbine
 - C. The centrifugal compressor
 - D. Bypass air at low altitude
2. If one igniter fails during engine start:
 - A. The engine starts normally
 - B. A hot start results
 - C. Combustion does not occur
 - D. The start sequence terminates automatically
3. Ignition during normal engine start is activated by:
 - A. Turning the ignition switches to ON at 10% N₂
 - B. Moving the throttle to IDLE at 10% N₂
 - C. Pressing the start button
 - D. The speed-sensing switch
4. Ignition during normal engine start is terminated by:
 - A. The speed-sensing switch on the starter-generator at 42%–48% N₂
 - B. Turning the ignition switch OFF
 - C. Turning the boost pump off
 - D. The fuel computer at 42%–48% N₂
5. Power is automatically applied to the igniters when the ignition switch is in NORM any time:
 - A. The start button is pressed during start and the throttle is out of IDLE CUT OFF
 - B. The wing anti-ice system is activated
 - C. The engine anti-ice switch is ON
 - D. Both A and C
6. Which statement about the TFE731 engine is correct:
 - A. The engine fuel system has a fuel heater, warmed by bleed air
 - B. The engine oil is cooled by fuel and air
 - C. The fan planetary reduction gears have a lubrication system independent of the engine system
 - D. The engine oil level should be checked within 30 minutes of engine shutdown
7. The LH–RH OIL PRESS WARN annunciator illuminates whenever the:
 - A. Oil temperature exceeds 127°C
 - B. Oil pressure is less than 25 psi
 - C. Oil filter bypasses oil
 - D. Fuel-oil cooler becomes clogged
8. Electronic fuel computer failure:
 - A. Is always indicated by illumination of the LH–RH FUEL COMP MANUAL annunciator
 - B. Is not possible with the switch in the NORM position
 - C. Is not always indicated by illumination of the LH–RH FUEL COMP MANUAL annunciator
 - D. Due to complete electrical failure, the LH–RH FUEL COMP MANUAL annunciator illuminates
9. If the fuel computer fails in flight, the surge bleed valve:
 - A. Opens fully
 - B. Closes when throttle angle is 42° or more
 - C. Assumes a 1/3-open position
 - D. Closes fully



10. Windmilling airstarts are not to be attempted with the fuel computer operating unless:
- A. Altitude is less than 20,000 feet
 - B. Stabilized N_1 rpm is less than 15%
 - C. Stabilized N_2 rpm is 15% or more
 - D. Altitude is above 2,000 feet
11. During normal ground start, the affected ignition light comes on:
- A. After pushing the starter button
 - B. When the throttle is moved to IDLE
 - C. At 20% N_2
 - D. At 42%–48% N_2
12. During a pre-takeoff fuel computer check, the N_2 of the right engine rapidly increases when the computer switch is turned to MAN. The following action is required:
- A. Leave the affected switch off for takeoff
 - B. Place the cockpit switch to NORM, and continue the flight with no further action
 - C. Turn the computer switch to NORM, shut the engine down, and have it checked before flight
 - D. Leave the switch off, and match the affected N_1 to the opposite engine for takeoff
13. The maximum transient oil temperature at altitude is:
- A. 149°C for two minutes
 - B. –40°C continuous
 - C. 127°C above 30,000 feet
 - D. 140°C below 30,000 feet

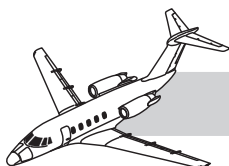


CHAPTER 8

FIRE PROTECTION

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CHAPTER 8

FIRE PROTECTION



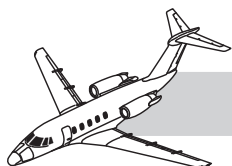
INTRODUCTION

The fire protection system for the Citation 650 series aircraft provides fire-detection and fire-extinguishing equipment for the engines and auxiliary power unit (APU). The fire detection system consists of three separate detection circuits that provide visual and aural warnings. Cabin smoke is detected by a sensor in the aft cabin, which sends a signal to illuminate an annunciator. The fire-extinguishing system consists of three fire bottles (two for the engines and one for the APU) that are activated from the cockpit. Two portable hand-held fire extinguishers also are in the aircraft.

GENERAL

The engine fire and overheat detection system consists of a detector/sensor, detection control unit, a FIRE warning light (Figure 8-1), and a FIRE warning bell. Circuit integrity is monitored by an amber FIRE DET FAIL annunciator.

On early models (SNs 0001—0151), the system requires 28 VDC from the left extension and right branch buses. On subsequent models, the system requires 28 VDC from the emergency branch DC bus. The system is tested by the rotary TEST knob on the center switch panel.



8 FIRE PROTECTION

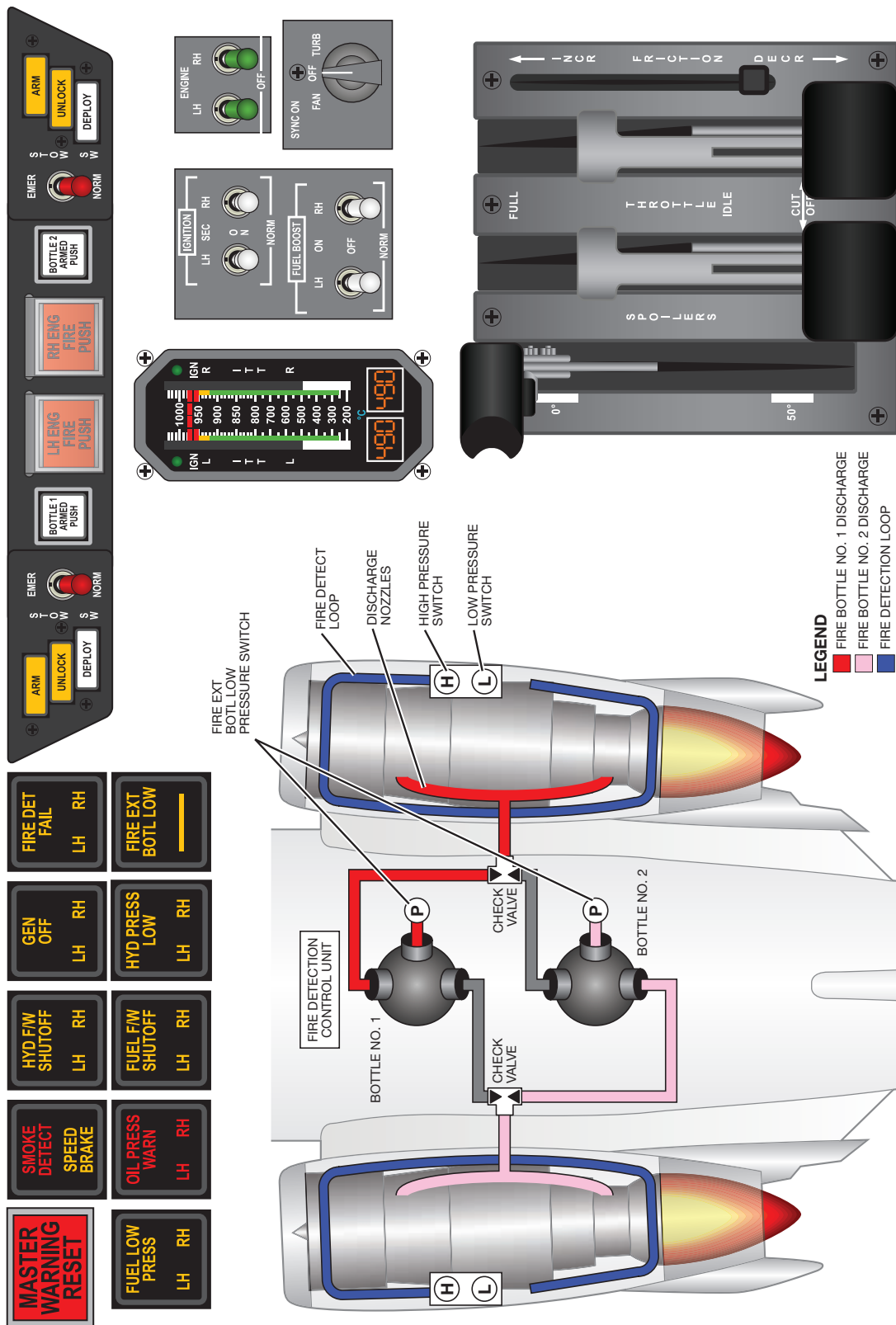


Figure 8-1. Engine Fire Detection System



The engine fire-extinguishing system consists of fire bottles charged with extinguishing agent, pressurized with dry nitrogen, and discharged by electrically activated squibs. The bottles are armed and activated manually from the cockpit. The bottles are guarded against overpressure. A FIRE EXT BOTTLE LOW annunciator illuminates when low pressure is detected in either fire bottle.

DESCRIPTION

ENGINE FIRE-DETECTION SYSTEM

The engine fire-detection system consists of:

- Engine fire sensors
- Detection control units
- ENG FIRE PUSH switchlights
- Fire bell
- Rotary TEST knob
- FIRE DET FAIL annunciator

The engine fire and overheat detection system indicates an overheat or fire condition.

CABIN SMOKE DETECTION

An optical sensor examines air leaving the cabin through the right outflow valve on the aft pressure bulkhead. If smoke is present, a red SMOKE DETECT annunciator (Figure 8-1) illuminates on the annunciator panel, and the MASTER WARNING RESET switchlights flash.

ENGINE FIRE-EXTINGUISHING SYSTEM

The engine fire-extinguishing system consists of two fire bottles, as well as deployment tubes, nozzles, and discharge control switches (Figure 8-1). The fire bottles are interconnected so that both bottles can be used for either engine (Figure 8-1). The fire bottles incorporate fill and pressure relief valves, temperature com-

pensating switches, and explosive cartridge operated discharge valves.

Each fire bottle contains a charge of CBrF_3 (bromotrifluoromethane), pressurized with dry nitrogen. The contents do not damage the engine or nacelle and require no cleanup or flushing. The extinguishing agent (containing no oxygen) extinguishes the fire by displacing the oxygen in the nacelle area.

If bottle pressure drops too low, the pressure switch closes and illuminates the amber FIRE EXT BOTTLE LOW annunciator (Figure 8-1). The bottle uses a combination fill fitting and safety relief valve. If the bottle temperature rises abnormally high, a valve in the fill fitting melts and relieves the bottle contents.

COMPONENTS

ENGINE FIRE SENSOR

Each engine is equipped with an engine fire sensor (Figure 8-1), which is a flexible stainless steel tube containing a fixed volume of inert gas. An increase in temperature anywhere on the tube increases the gas pressure. A fire-detection control unit containing two pressure switches is connected to the end of each tube. One switch is for the alarm and the other is for integrity.

When a fire or overheat condition increases the gas pressure enough to close the alarm switch, an electrical signal is sent to the fire-detection control unit. The signal illuminates the red LH–RH ENG FIRE PUSH switchlight and sounds the warning bell for one 3-second interval.

NOTE

Illumination of the LH or RH ENG FIRE PUSH switchlight does not activate the MASTER WARNING RESET switchlights.

When the pressure decreases, the alarm switch opens and deactivates the LH or RH ENG



FIRE PUSH switchlight. The integrity switch is normally closed. If the inert gas escapes from the tube, the integrity switch opens and illuminates the **FIRE DET FAIL** annunciator (Figure 8-1).

CABIN FIRE EXTINGUISHERS

The aircraft is equipped with two portable hand-held fire extinguishers. One is in the passenger compartment, with its location dependent on interior design. The other is in the cockpit under the copilot seat (Figure 8-2). The extinguishers are suitable for use on Class A, B, and C fires.



Figure 8-2. Cockpit Fire Extinguisher

of the cockpit glareshield (Figure 8-1). The switchlights illuminate when the fire bottles are armed to release extinguishing agent. After the extinguishing agent is released, the respective annunciator extinguishes and is no longer available for use.

ROTARY TEST KNOB

A rotary **TEST** knob is on the center switch panel (Figure 8-3), and is used for testing the fire protection system.

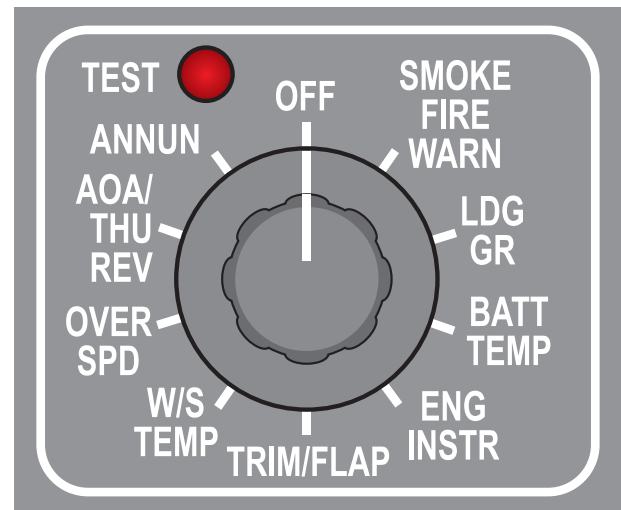


Figure 8-3. Rotary TEST Knob

CONTROLS AND INDICATIONS

For information on the annunciators associated with the engine fire detection system refer to Chapter 4—"Master Warning System."

ENG FIRE PUSH SWITCHLIGHTS

The guarded red LH and RH **ENG FIRE PUSH** switchlights (Figure 8-1) are in the center of the cockpit glareshield.

BOTTLE ARMED PUSH SWITCHLIGHTS

The **BOTTLE 1 ARMED PUSH** and **BOTTLE 2 ARMED PUSH** switchlights are on the center

OPERATION

PREFLIGHT

Testing the Firewall Shutoff Valves

When the ground temperature is -15°C ($+5^{\circ}\text{F}$) or below, the firewall shutoff valves must be exercised prior to the Starting Engines Checklist. To do so, press the **ENG FIRE PUSH** switchlight, and then verify that the respective **FUEL F/W SHUTOFF** and **HYD F/W SHUTOFF** annunciators and both **BOTTLE ARMED PUSH** switchlights illuminate.



Testing the Fire Protection System

To test the fire protection system, select the SMOKE FIRE WARN position on the rotary TEST knob, and then verify illumination of the LH and RH ENG FIRE PUSH switchlights and the LH and RH FIRE DET FAIL annunciators. Also verify that the fire bell sounds for a single, 3-second interval. The MASTER WARNING RESET switchlights illuminate when the red SMOKE DETECT annunciator is activated by the rotary TEST knob.

ENG FIRE PUSH SWITCHLIGHTS

If a fire is indicated, in conjunction with memory action items, lift the cover for the illuminated ENG FIRE PUSH switchlight, and then press the switchlight. When the switchlight is pressed, the following occurs:

- The fuel and hydraulic firewall shutoff valves are closed
- The field relay on the generator is tripped
- Both fire bottles are armed
- The thrust reverser is disabled for normal operation

Firewall shutoff and extinguisher arming are indicated by the illumination of the following annunciators:

- FUEL F/W SHUTOFF LH–RH
- HYD F/W SHUTOFF LH–RH
- GEN OFF LH–RH
- BOTTLE 1 ARMED PUSH/BOTTLE 2 ARMED PUSH (both fire bottles)

After activating the ENG FIRE PUSH switchlight, pressing the switch a second time reverses the actions and extinguishes all annunciators except the GEN OFF annunciator. The generator must be reset.

CAUTION

After the ENG FIRE PUSH switchlights are pressed, the white BOTTLE 1 ARMED PUSH and BOTTLE 2 ARMED PUSH switchlights illuminate and the circuit to the fire bottle cartridges is complete. If either BOTTLE ARMED PUSH switchlight is subsequently pressed, then the bottle is discharged and must be recharged or replaced prior to flight.

EXTINGUISHING A FIRE

If a fire is indicated, then the respective LH and/or RH ENG FIRE PUSH switchlight illuminates.

To activate the fire extinguisher system, raise the plastic guard and then press the respective ENG FIRE PUSH switchlight. Doing so illuminates the BOTTLE 1 AND BOTTLE 2 ARMED PUSH switchlights and provides electrical power to the discharge switches.

Pressing either BOTTLE ARMED PUSH switchlight discharges that bottle's fire extinguishing agent into the engine nacelle. The respective annunciator extinguishes indicating that the circuit is activated.

When a bottle is discharged, the FIRE EXT BOTTLE LOW annunciator illuminates. If the ENG FIRE PUSH switchlight remains illuminated after the first fire bottle discharges, then fire is still present.

NOTE

Perform the Engine Fire Checklist prior to pushing the remaining BOTTLE ARMED PUSH switchlight.

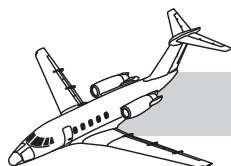
PORTABLE FIRE BOTTLES

Illumination of the red SMOKE DETECT annunciator indicates the presence of smoke. If a cabin fire is present, unlatch the portable extinguisher from its support, and then discharge the extinguishing agent by pulling the pin and squeezing the spring-loaded lever. The lever can be released any time to stop the discharge.



QUESTIONS

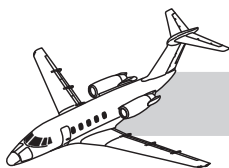
1. Pressure in either engine fire extinguisher bottle has dropped too low for use. The indication that is seen is:
 - A. Illumination of the FIRE EXT BOTTLE LOW annunciator
 - B. A frangible blowout disc is missing
 - C. Illumination of the FIRE DET FAIL LH–RH annunciator
 - D. A pop-out warning button is visible on the fire extinguisher bottle in the tailcone
2. An engine fire-detection tube is broken. The cockpit indication is:
 - A. Illumination of the ENG FIRE PUSH switchlight
 - B. Illumination of the FIRE EXT FAIL annunciator
 - C. Illumination of the FIRE DET FAIL annunciator
 - D. Illumination of the APU FAIL annunciator
3. Illumination of the ENG FIRE PUSH switchlight is accompanied by:
 - A. Illumination of a FIRE BOTTLE ARMED switchlight
 - B. Ringing of the fire bell for 3 seconds
 - C. Illumination of the FIRE DET FAIL switchlight
 - D. Illumination of the MASTER WARNING RESET switchlights
4. When the fire-extinguishing system is armed for operation:
 - A. The FUEL LOW PRESS annunciator illuminates
 - B. The HYD PRESS LOW annunciator illuminates
 - C. The GEN OFF annunciator illuminates
 - D. All the above
5. If the contents of a bottle have been discharged into a nacelle and the ENG FIRE PUSH switchlight remains on:
 - A. Fire has been extinguished
 - B. The other bottle can be discharged into the same nacelle by pressing the other BOTTLE ARMED PUSH switchlight
 - C. The fire still exists, but no further action can be taken
 - D. The same BOTTLE ARMED PUSH switchlight can be pressed again, firing a second charge of agent from the same bottle
6. The firewall shutoff valves can be reopened:
 - A. With the MASTER CAUTION switchlight
 - B. By pressing the ENG FIRE PUSH switchlight
 - C. By selecting the SMOKE/FIRE DETECT position on the rotary TEST knob
 - D. Only by maintenance personnel



CHAPTER 9 PNEUMATICS

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CHAPTER 9

PNEUMATICS



INTRODUCTION

The pneumatic system for the Citation 650 series uses engine compressor bleed air for various systems. Pneumatic air can be routed from various sources to include both engines or an auxiliary power unit (APU). During single-engine operation, air from one engine is sufficient to maintain all required system functions.

GENERAL

High and low pressure bleed air from each engine is routed through check valves into the bleed air system. Bleed air is used for the pneumatic air conditioning (PACs), anti-ice systems, service air, rudder bias actuator, and emergency pressurization. The service air sys-

tem supplies regulated air for the door seal and an air ejector for generating vacuum for pressurization control. The PACs use high or low pressure bleed air. All other components use high pressure bleed air.



DESCRIPTION

BLEED AIR DISTRIBUTION

Low pressure bleed air is routed through a limiting venturi and a check valve into the bleed air manifold (Figure 9-1 and 9-2). High pressure air is routed through a bypass valve, a pre-cooler, and a check valve to all systems it supplies (bypass valve and check valve not installed on SNs 7001—7119).

High pressure bleed air is routed through the heat-exchanger assembly where a temperature sensor connected to the HP PRECLER O'HEAT LH-RH annunciator senses overtemperature. The air is then routed through control valves to the nacelle anti-ice system. Air also is routed through check valves and a pressure regulating valve to the vacuum ejector and through another check valve to the main entrance door seal.

Before the air enters the service air check valve, it is extracted and routed to the rudder bias actuator, the PAC HP valves, and the anti-ice/emergency pressurization shutoff valves. If the PAC HP valve is open, a pressure switch causes the appropriate PAC HP VLV OPEN LH-RH light to illuminate. When the anti-ice/emergency pressurization shutoff valve is open, air is routed through check valves to the windshield anti-ice valve, the wing anti-ice valves, and the emergency pressurization valves.

COMPONENTS

RUDDER BIAS SYSTEM

High pressure bleed air for operating the rudder bias system is extracted from the left and right engines. The bleed air is extracted from the service air supply before reaching the service air pressure regulator (Figure 9-1 and 9-2). High pressure bleed air is routed to the left and right inlets on the rudder bias actuator. For details on rudder bias operation, refer to Chapter 15—"Flight Controls."

SERVICE AIR SYSTEM

Service air furnishes regulated air pressure for inflating the passenger entry door seal and for operating an air ejector.

The pressure regulator controls air pressure and a relief valve in the regulator prevents outlet pressure from exceeding limits if the regulator malfunctions.

Regulated air is routed through the ejector, which creates vacuum for cabin pressurization control. The air is also routed forward to the aft door frame of the passenger entry door where the door seal valve is located.

Service air enters the door seal through the lower door frame, both halves of which mate when the cabin entry door is closed. When the entry door is locked, the lower aft door latch pin depresses the seal inflation pneumatic valve, allowing service air to flow into the inflatable seal. A pressure switch on the door seal actuates the DOOR SEAL annunciator if the door seal loses inflation pressure. When the door seal is inflated, the pressure switch extinguishes the annunciator.

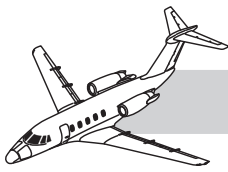
EMERGENCY PRESSURIZATION SYSTEM

Emergency pressurization is activated automatically by a barometric pressure switch at a cabin altitude of approximately 13,500 feet. If the cabin altitude descends below 13,500 feet, then emergency pressurization shuts off automatically.

CONTROLS AND INDICATIONS

PAC BLD SELECT SWITCH

High pressure air enters the bleed-air manifold through a valve controlled by the PAC BLD SELECT switch (Figure 9-1 and 9-2).



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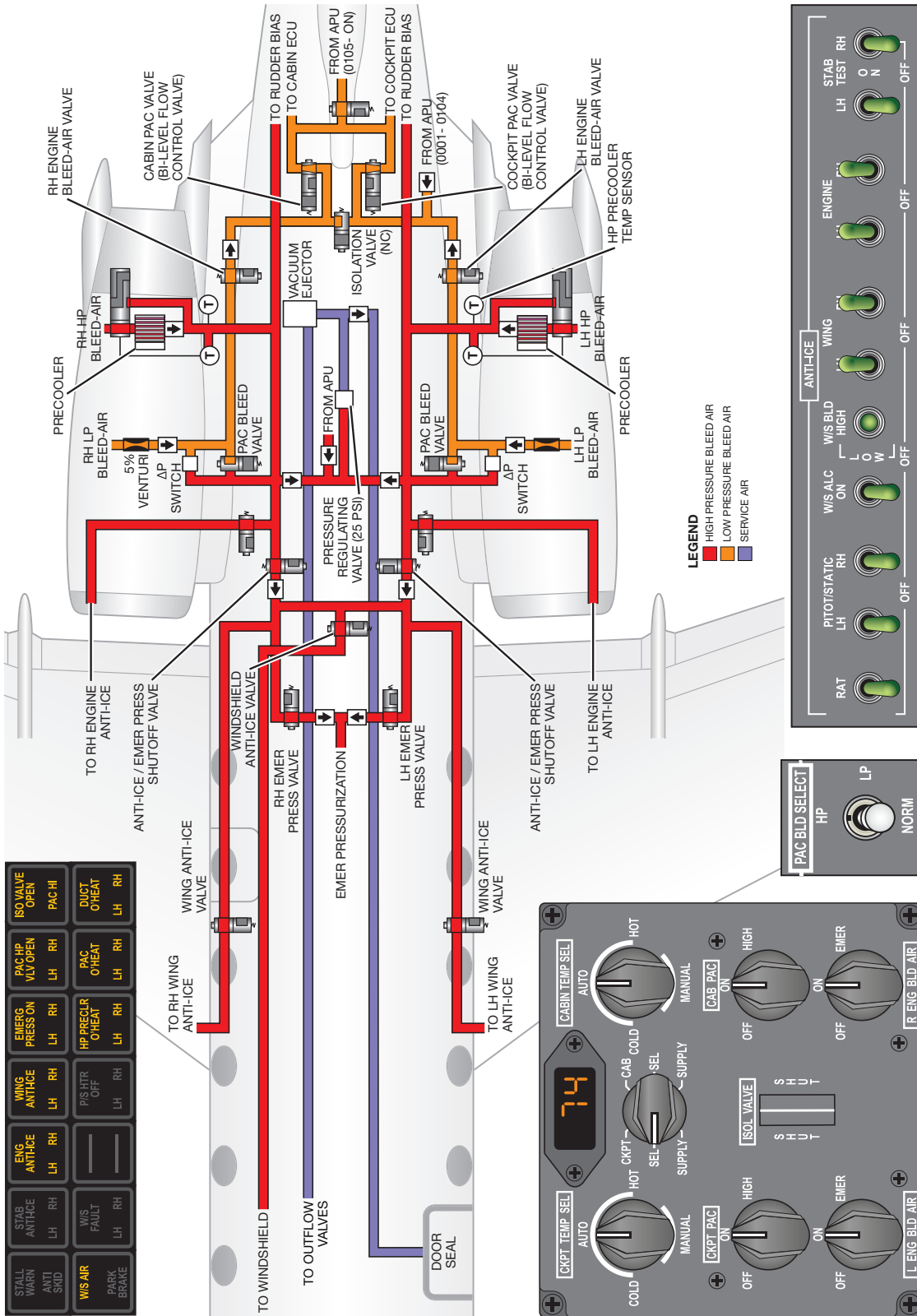


Figure 9-1. Pneumatic System: SNs 0001 – 0241

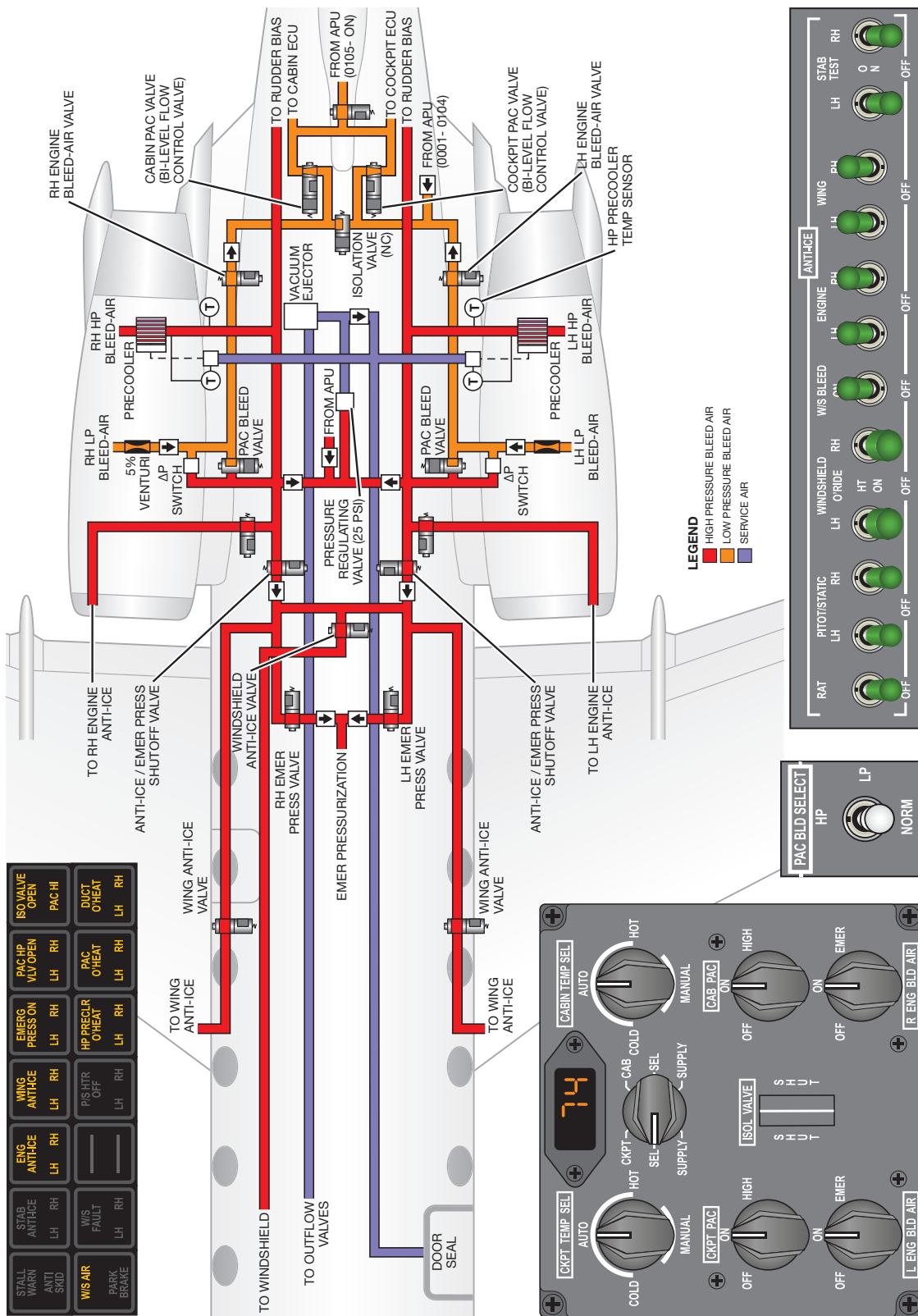


Figure 9-2. Pneumatic System: SNs 7001–7119



The PAC BLD SELECT switch has three positions: HP (high pressure), LP (low pressure) and NORM (normal). In the HP position, the PACs receive only high pressure bleed air. In the LP position, PACs receive only low pressure bleed air. In the NORM position, high pressure bleed air is used whenever the throttles are below 55% N_1 throttle switch position (determined on the ground).

The exception is during flight with the landing gear extended. Above 55% N_1 , the PAC HP valves are closed, allowing only low pressure bleed air to the PACs. Normally, the right engine supplies bleed air to the cabin PAC, and the left supplies the cockpit PAC. An isolation valve joins the two systems and permits one engine to supply both PACs or both engines to supply one PAC. Both engines supply all other systems requiring bleed air for operation.

ENGINE BLEED AIR SWITCH

The ENG BLD AIR switch (Figure 9-1 and 9-2) has three positions: OFF, ON, and EMER. The switch controls the solenoid regulating and shutoff valve. The ON position de-energizes the valve open and the OFF position energizes the valve closed.

When the switch is in the EMER position, the appropriate EMERG PRESS ON LH–RH annunciator illuminates. When EMER is selected the anti-ice/emergency pressurization shutoff valve opens, as does the emergency pressurization valve. The open valves supply HP bleed air directly to the cabin, bypassing the PACs. The shutoff and regulating valve remains open, allowing air from the engine to go to the PACs.

OPERATION

The bleed-air system operates automatically. Normal crew responsibility for operating the system is limited to selection of the bleed air and PAC air source.

When emergency pressurization is activated or selected, the appropriate EMERG PRESS ON LH–RH annunciator illuminates and increased noise can be heard. To shut off emergency pressurization after automatic activation, place the appropriate ENG BLD AIR selector to EMER and then back to ON or OFF.

LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

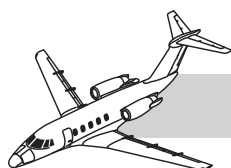
EMERGENCY/ ABNORMAL

For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.



QUESTIONS

1. Systems that use pneumatic bleed air for operation are:
 - A. Nacelle anti-ice, emergency brakes, and the entrance door
 - B. Wing, windshield, and nacelle anti-ice, rudder bias, main door seal, PACs, and vacuum ejector
 - C. Entrance door, rudder bias, instrument air, and PACs
 - D. Emergency pressurization and PACs only
2. Emergency pressurization air:
 - A. Is supplied only from the left engine
 - B. Activates automatically when cabin altitude reaches 13,500 feet
 - C. When selected prevents bleed air from going to the associated environmental control unit
 - D. Activates on automatically if an ECU overheats

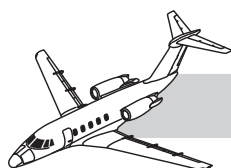


CHAPTER 10

ICE AND RAIN PROTECTION

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CHAPTER 10

ICE AND RAIN PROTECTION



INTRODUCTION

The Citation 650 series aircraft is equipped with ice and rain protection systems. The aircraft is approved for flight into known icing conditions when the required equipment is installed and functioning properly. Ice protected areas include the wing and horizontal stabilizer leading edges, engine inlets, windshield, pitot tubes, static ports, and angle-of-attack probes. These are preventative or anti-icing systems which should be checked prior to flight when icing conditions are anticipated and activated prior to entering icing conditions.

GENERAL

The aircraft are equipped with ice detection and anti-ice systems, and a windshield defog and rain removal system. Ice and rain protec-

tion switches are green for easy identification. Ice detection systems include windshield ice detection lights and wing inspection lights.



The anti-ice systems include:

- Pitot and static anti-ice system
- Rudder bias heating blanket system
- RAM air temperature anti-ice system
- Windshield anti-ice system
- Engine anti-ice system
- Wing anti-ice system
- Horizontal stabilizer anti-ice system
- Heated drains

Ice formation is prevented through the use of electrically heated elements on the following components:

- Pitot tubes
- Static ports
- Rudder bias heat blanket
- Ram air temperature (RAT) probe

- Engine P_{T2}/T_{T2} probe
- Wing root fairing
- Generator air inlets
- Horizontal stabilizer leading edges
- Windshields (VII only)

The relief tube and refreshment center drains are also electrically heated to prevent ice formation.

Ice formation is prevented through the use of high pressure bleed air on the following components:

- Windshields (Citation III/VI) (Citation VII rain removal only)
- Wing leading edges
- Leading edge of the engine nacelle

Figure 10-1 shows the ice protected surfaces of the aircraft. Figure 10-2 shows the ice and rain controls and indications for the aircraft.

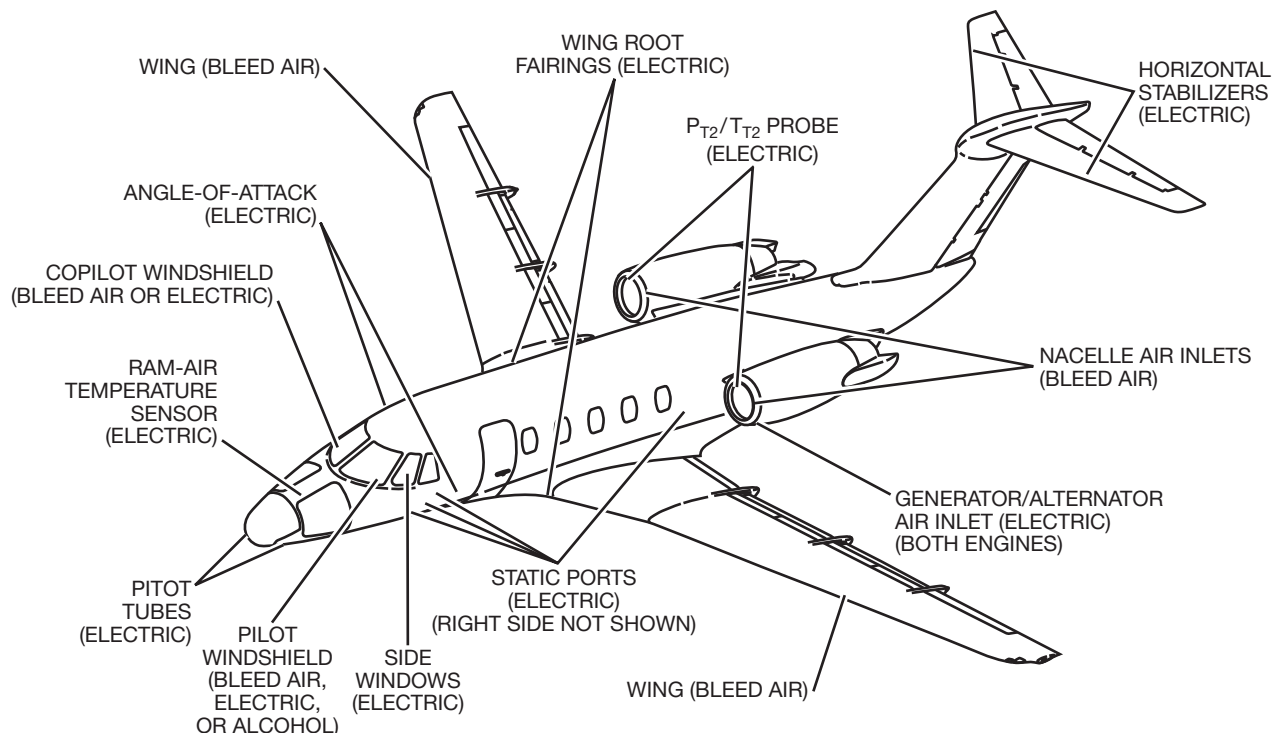


Figure 10-1. Ice-Protected Surfaces

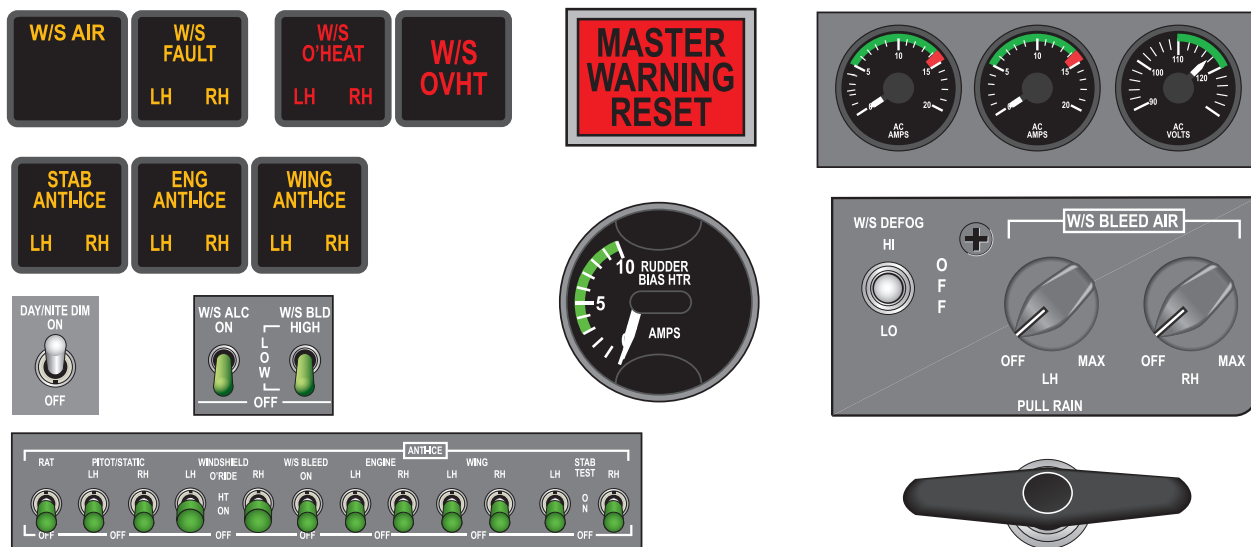
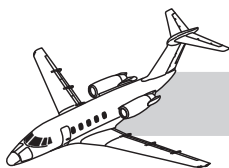


Figure 10-2. Ice and Rain Controls and Indications

ICE DETECTION SYSTEMS

WINDSHIELD ICE DETECTION LIGHTS

The windshield ice detection lights on the glareshield allow ice detection in dark conditions. These lights show ice accumulation on the windshield by reflecting a circular red glow on the pilot and copilot windshields. The pilot windshield ice detection light is near the center of the pilot windshield. The copilot ice detection light is located near the RH inboard element (Figure 10-3).

To illuminate or extinguish the windshield ice detection lights place the DAY/NITE DIM switch (Figure 10-2) in the ON or OFF position.

To verify normal operation of the ice detection lights place the DAY/NITE DIM switch to the ON position and then place the palm of the hand over the lights. A red glow indicates that the lights are operating normally.

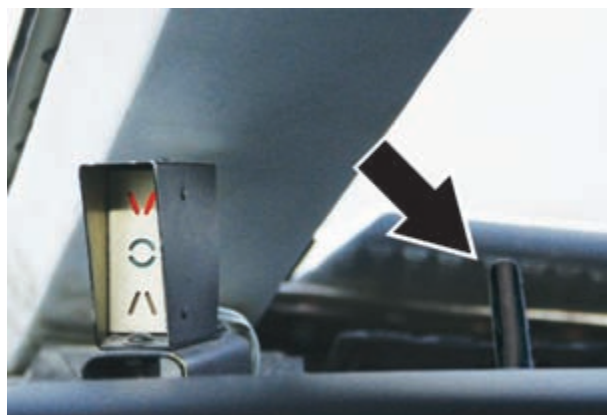


Figure 10-3. Windshield Ice Detection Light

NOTE

When the ice detection lights indicate icing, all anti-ice systems must be activated.

WING INSPECTION LIGHTS

The wing inspection lights near each wing allow detection of ice accumulation in dark conditions (Figure 10-4). To illuminate the wing inspection lights, place the switch labeled OFF-WING INSP-RECOG/TAXI to the WING INSP position.



ANTI-ICE SYSTEMS

PITOT AND STATIC ANTI-ICE SYSTEM

The pitot/static anti-ice system prevents ice formation on the pitot tubes and static ports.

The PITOT/STATIC switches control electrical power from the left extension or emergency branch and right branch buses to the following components (Figure 10-5):

- Two pitot tube heaters
- Six flight instrument static port heaters
- Two static port heaters for the outflow valves
- Two angle-of-attack probe heaters
- RH PITOT STATIC switch activates rudder bias heat

A failure in the system is indicated by illumination of the P/S HTR OFF LH–RH annunciator. Current sensors illuminate the annunciator if a failure occurs in a pitot tube or static port. The annunciator also illuminates when the switch is OFF.

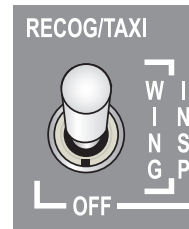


Figure 10-4. Wing Inspection Control Switch and Light



Figure 10-5. Pitot-Static Anti-Ice



RUDDER BIAS HEATING BLANKET SYSTEM

The rudder bias heating blanket system prevents ice formation on the rudder bias actuator. The RH PITOT/STATIC switch provides electrical power for normal operation of the heating element. When the switch is placed in the ON position, a thermal switch in the blanket cycles the power on and off as required to maintain the temperature. When the power cycles, the rudder bias ammeter needle moves accordingly.

The RUD BIAS switch, a three-position switch on the center switch panel (Figure 10-6), provides electrical power for testing the rudder bias heating blanket system. The heat blanket is powered by completion of a circuit from the RUDDER BIAS HTR circuit breaker on the RH feed bus.

To test the system, place the RUD BIAS switch in the TEST position and observe the rudder bias ammeter. A minimum rise of 7 amps on the ammeter indicates proper system operation (Figure 10-6).

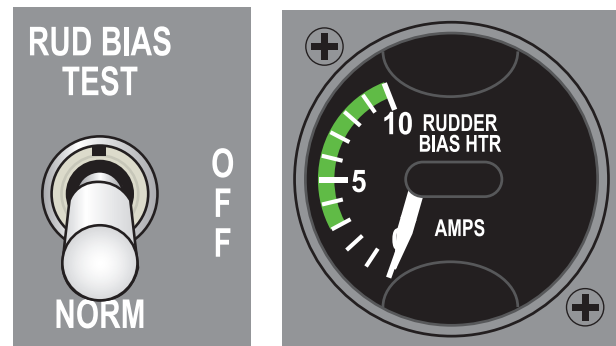
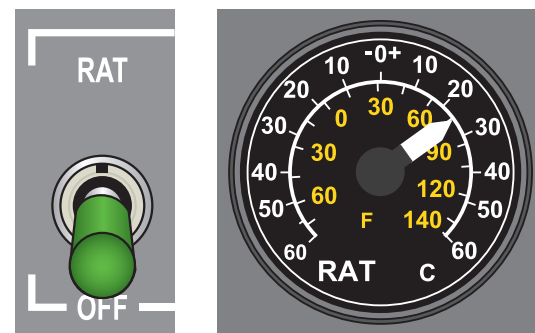


Figure 10-6. Rudder Bias Heat Blanket Controls and Indications



RAM AIR TEMPERATURE ANTI-ICE SYSTEM

The ram-air temperature system probe is electrically heated through the RAT switch on the center switch panel (Figure 10-7). To check the system place the RAT switch to ON. A rise on the RAM AIR temperature gauge indicates proper system operation.

BLEED AIR WINDSHIELD ANTI-ICE/RAIN REMOVAL SYSTEM

Description

The windshield bleed air system of each aircraft model operates in the same manner except for the air control valve and the bleed air switch. High pressure bleed air is used for either windshield anti-ice or rain removal. The W/S BLD or W/S BLEED switch is on the center switch panel (Figures 10-8 and 10-9).

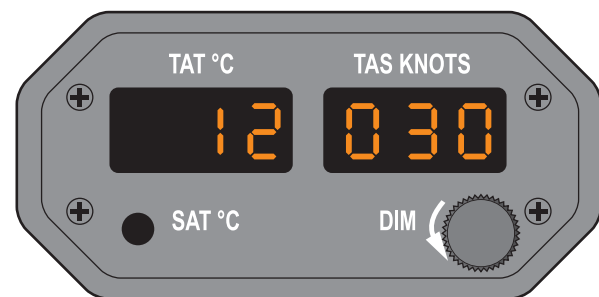


Figure 10-7. RAT Switch and RAM AIR Temperature Gauge

Depending on the aircraft, the switch has either two or three positions. Citation III/VI aircraft have a three position switch; OFF–LOW–HIGH. Citation VII aircraft have a two position switch, OFF–ON and have no air control valve. The system is primarily for rain removal on the Citation VII.





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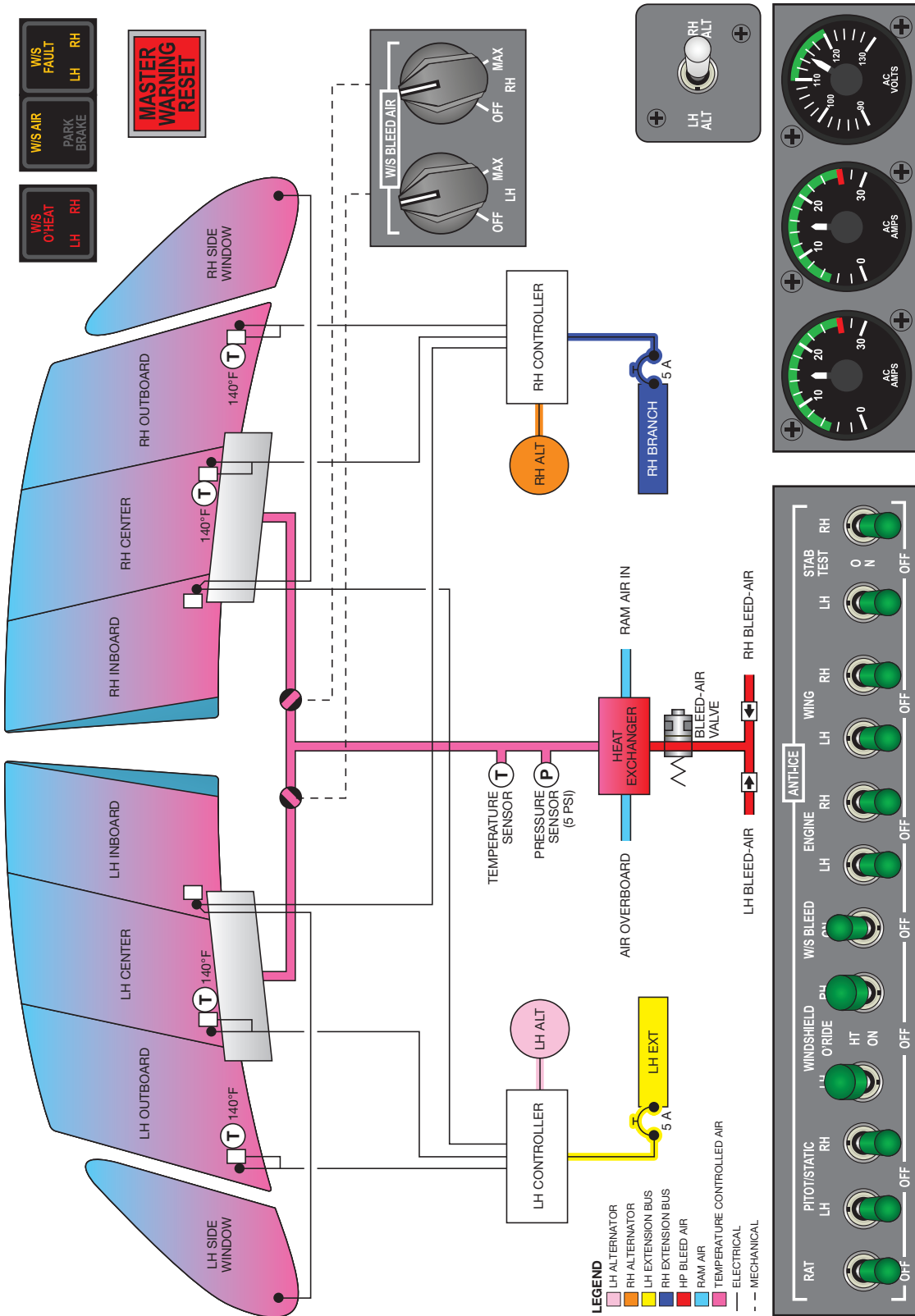


Figure 10-9. Windshield Anti-Ice—Citation VII



Airflow is modulated by two valves controlled by the W/S BLEED AIR knobs on the copilot lower instrument panel.

Activating the W/S BLD switch opens a bleed air valve (and also activates a temperature controller in the Citation III/VI), allowing hot, high pressure engine bleed air into the bleed air duct. The duct runs through a tail-mounted heat exchanger, which uses ram air to cool the windshield bleed air.

The ram air passes through the heat exchanger and is exhausted overboard behind the baggage compartment door. From the heat exchanger, the bleed air flows through the duct to manual valves in the nose section.

The temperature controller in the Citation III/VI receives the following inputs:

- W/S BLD switch position
- Air temperature going to the windshield (nose sensor)
- Air temperature in the bleed air duct (tail sensor)

Normal system operation is indicated by an increase in the noise level as the bleed air discharges through the nozzles.

Depending on the aircraft, a temperature sensor in the bleed air duct causes automatic illumination of either the W/S AIR or the W/S O'HEAT annunciator and MASTER WARNING RESET switchlights when the bleed air temperature is too high. During overtemperature conditions, the bleed air valve closes, stopping bleed air flow to the windshield. Overtemperature conditions may occur if a sustained high power, low airspeed condition is maintained or if a system malfunction occurs. When the system returns to a normal temperature, the temperature sensor automatically reopens the bleed air valve, and the W/S AIR and/or W/S O'HEAT annunciator will extinguish, however the MASTER WARNING RESET switchlights will continue to flash until manually reset.

NOTE

The system is designed to prevent illumination of the W/S O'HEAT LH–RH annunciator if windshield bleed air is used during takeoff. If the W/S BLD switch is ON when the aircraft is on the ground and the throttles are advanced beyond 75% N_1 , then illumination of the O'HEAT warning light is delayed for 30 seconds. If the aircraft is still on the ground after 30 seconds and bleed air temperature exceeds predetermined limits, then the W/S O'HEAT LH–RH and MASTER WARNING RESET switchlights illuminate.

If the W/S AIR or the W/S O'HEAT annunciator cycles on and off, indicating an overheat condition when the W/S BLD switch is activated, then partially close the W/S BLEED AIR knob to reduce bleed air flow.

When the W/S BLD switch is OFF and the W/S AIR annunciator illuminates (steady light), then a malfunction of the bleed air valve is indicated.

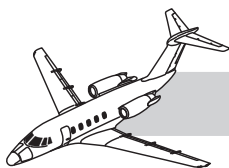
If an electrical power loss occurs, then the solenoid bleed air valve fails open, allowing non-temperature controlled engine bleed air to flow to the windshield if the manual valves are open.

NOTE

The aircraft are normally flown with the W/S BLEED AIR knobs in the OFF (closed) position. Open the knobs (positioned toward MAX) only when the windshield bleed air system is used.

The procedure mentioned in the previous note protects the windshield from inadvertent application of hot engine bleed air if electrical power is lost or if the solenoid valve fails.

To test the windshield bleed air system place the rotary TEST knob to W/S TEMP. The warning circuit is verified by illumination of the



W/S AIR and W/S O'HEAT annunciators and MASTER WARNING RESET switchlights. The W/S BLEED switch must be activated for the test to occur.

Rain Removal

The rain removal system uses normal windshield bleed air, along with rain doors that deflect airflow over each windshield. The rain doors are operated manually by pulling the PULL RAIN handle under the copilot panel (Figure 10-10). For rain removal, pull the RAIN handle out, place the W/S BLEED AIR knobs to the MAX position and place the W/S BLD switch to the LOW or ON position.



Figure 10-10. PULL RAIN Handle and Rain Removal Doors

NOTE

Opening the rain door may be difficult if the windshield bleed air system is operating.

To increase airflow to the pilot side windshield during periods of low power settings, such as during the landing approach, set the copilot side W/S BLEED AIR knob to the OFF position, which diverts all available bleed air to the pilot side windshield.

ALCOHOL SYSTEM (CITATION III/VI ONLY)

Windshield alcohol anti-ice is used as backup to the bleed air. (Figure 10-11). The system consists of an alcohol reservoir, pump, and nozzles to provide up to a 15-minute supply of alcohol to the pilot windshield only. The alcohol reservoir is checked prior to flight through a sight gauge in the left nose com-

partment. The reservoir contains 2 quarts of isopropyl alcohol lasting fifteen minutes.



Figure 10-11. Windshield Alcohol Reservoir and Anti-Ice Nozzles

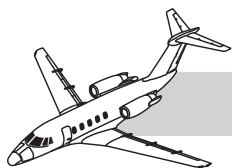
The windshield alcohol is controlled by a two-position W/S ALC switch on the center switch panel. When the W/S ALC switch is placed in the ON position, electrical power from the left extension bus is sent to the pump, which delivers alcohol to the alcohol nozzles on the pilot windshield.

ELECTRIC WINDSHIELD ANTI-ICE SYSTEM (CITATION VII ONLY)

Description

Each windshield and forward side window is manufactured from herculite glass by Pittsburg Paint and Glass (PPG). Each windshield incorporates three electrical heating elements. Figure 10-9 shows the electric windshield anti-ice system for the Citation VII. The elements are configured vertically and are considered inboard, center, and outboard.

Each heating element contains a sensor (Figure 10-9). The outboard and center elements



contain the primary and secondary sensors respectively, and these sensors are active. The inboard elements contain a spare sensor, which is not used in normal operation.

Power is supplied by two 7.5 KVA alternators (one on each engine). The system incorporates left and right electric windshield controllers and left and right alternator power control units.

The outboard and center elements on each windshield are powered by their same-side alternators. The inboard element of each windshield is powered by the opposite side alternator. The left windshield inboard element is powered by the right engine alternator. The right inboard element by the left engine alternator.

The side windows contain no sensors. They are powered in parallel from the inboard power elements on their respective sides.

If the left side temperature controller or alternator fails, then a minimum of 1/3 (inboard element) of the left windshield surface remains heated from the operating right side alternator, and 2/3 of the right windshield surface (outboard and center elements) remains heated. The pilot side window will be heated, but the copilot side window will not.

If the right side alternator or controller fails, then a minimum of 2/3 of the left windshield (center and outboard) surface remains heated and 1/3 of right windshield (inboard only) remains heated. The right side window will be defogged, and the left side window will not.

Test the windshield anti-ice system prior to flight with the aircraft engines running, which allows the alternator to provide the AC power necessary to test the system.

With the engines running, place the windshield heat switches (Figure 10-12) to the HT ON position, and then place the rotary TEST knob to the W/S TEMP position. A satisfactory test is indicated by the W/S FAULT LH–RH and W/S O’HEAT LH–RH annunciators illuminating and then extinguishing within approximately 2 seconds. The MASTER WARNING

RESET switchlights also illuminates during the test.

Turn off the TEST switch and all windshield switches. If either or both W/S FAULT annunciators remain illuminated, then the controller has detected system faults, such as phase imbalance, sensor faults, or built-in test failure.

If either or both W/S O’HEAT annunciators remain illuminated, then a failure of the overheat warning system is indicated. Any failure indicated during the preflight test must be corrected prior to flight.

Operation

To use the windshield anti-ice system, place the windshield heat switches (Figure 10-12) to either the HT ON or O’RIDE positions. The switches are lever lock switches and require positive action to move them to either OFF–HT ON–O’RIDE positions.

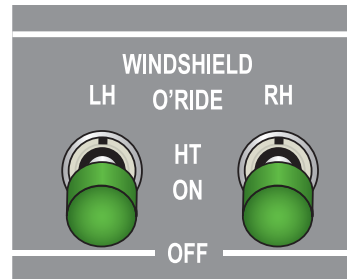


Figure 10-12. Windshield Heat Switches—Citation VII

When the switch is placed to the HT ON position, power is applied gradually to prevent thermally shocking a cold-soaked windshield. If, however, excessive windshield icing is encountered unexpectedly, place the switch to the O’RIDE position. This position provides maximum electrical power to the windshield in the least amount of time. The system cycles normally after reaching proper control temperatures. The electrical anti-icing system is also used for defogging, either alone or with the defog fan.



If, while the system is operating normally, the W/S O'HEAT LH–RH annunciator illuminates, then a temperature greater than preset limits on the affected windshield is indicated and the system shuts down.

The W/S O'HEAT annunciator extinguishes as the temperature decreases, and electrical power is reapplied automatically to the windshield.

If a system failure such as loss of a sensor or windshield controller, or a phase imbalance occurs, then the W/S FAULT LH–RH annunciator illuminates.

The AC ammeter gauges on the copilot meter panel (Figure 10-13) cycle when the windshield heat is operating. When the windshield anti-ice system is started, the gauges fluctuate until the windshields are at normal operating temperature, especially if the stabilizer heat is also operating. These fluctuations are considered normal.

When the windshield anti-ice system is activated, the alternator field relays are reset, indicated by approximately 115 VAC on the AC voltmeter (Figure 10-13). To check individual alternator voltage, use the AC voltmeter selector switch next to the AC voltmeter gauge.

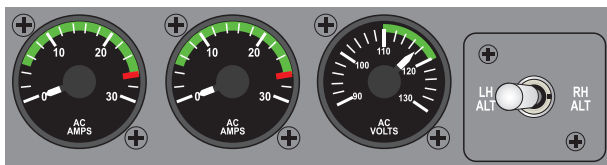


Figure 10-13. A/C Meter Panel—Citation VII

ENGINE ANTI-ICING

Description

The engine anti-ice system (Figure 10-14) consists of an engine nacelle lip heated by engine bleed air, an electrically heated generator/alternator air inlet, P_{T2}/T_{T2} probe, and the inboard wing root fairing.

Bleed Air

With the engines running, turning the ENGINE ANTI-ICE switches on (Figure 10-14) removes electrical power from the solenoid-operated anti-ice valves, which open and allow high pressure bleed air to flow into the engine nacelle lip. This air heats the nacelle lip and exits through four louvers. The ENG ANTI-ICE LH–RH annunciator illuminates when the respective ENGINE ANTI-ICE switch is on and the temperature of the nacelle lip is less than set limits. The temperature of the bleed air to the lip is controlled by the engine precooler.

Electrical

Generator/Alternator Air Inlet

The nacelle air inlet for generator and alternator cooling (Figure 10-15) is electrically heated by DC power from the left and right feed buses when the ENGINE LH–RH switches are placed in the on position. The heating elements are not monitored by the ENG ANTI-ICE annunciator; therefore, if a failure occurs in the system, the respective ENG ANTI-ICE annunciator does not illuminate.

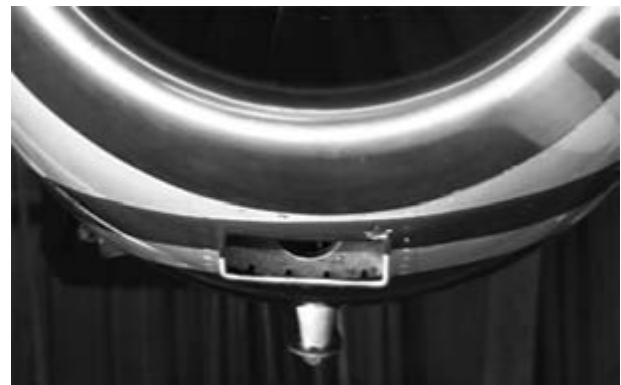


Figure 10-15. Generator/Alternator Air Inlet

P_{T2}/T_{T2}

The engine inlet pressure/temperature probe is also electrically heated when the ENGINE LH–RH switch is placed in the on position (Figure 10-16). Failure of the probe heater to draw normal current causes the appropriate ENG ANTI-ICE annunciator to illuminate.





Figure 10-16. PT₂/T₂ Probe



Figure 10-17. Electrically Heated Wing Root Fairing

Wing Root Fairing

The wing root fairing is electrically heated by 28 VDC power from the left and right feed buses when the engine anti-ice system is in use (Figure 10-17). A thermal blanket, bonded inside the fairing, heats on a periodic basis and is sequenced by a temperature controller. A temperature sensor molded into the thermal blanket supplies the temperature controller with a predetermined range of settings. An overheat condition is monitored by a thermoswitch, also molded into the blanket, which interrupts the current flow to the heater blanket until the overheat condition has cooled. Low temperature is also monitored by a thermoswitch. The ENG ANTI/ICE annunciator illuminates for either extreme condition.

CAUTION

The engine anti-ice system is designed as an anti-ice system. Its use should be anticipated and the system activated anytime flight in visible moisture with a RAT below +10°C (50°F) is imminent. Failure to turn on the system before ice accumulation has begun may result in engine damage due to ice ingestion.

Indications

Turning the engine anti-ice system on activates the ignition system and illuminates the ignition lights at the top of each interstage turbine temperature (ITT) indicator. The power source for the ignition system is supplied from the left and right feed buses when the engine

anti-ice switches are turned on. Turning the system on also illuminates the ENG ANTI-ICE LH–RH annunciator momentarily, and causes an increase on the ammeters. The ENG ANTI-ICE LH–RH annunciator remains illuminated or reilluminates if any of the following conditions exist:

- Nacelle lip temperature is below preset limits.
- Wing root fairing temperature is above the normal control range, indicating a failure of the temperature controller. This is a cycling light which extinguishes when the temperature drops below set limits.
- Wing root fairing temperature is below set limits.
- PT₂/T₂ probe heater fails.

WING ANTI-ICING

The left and right wing leading edges are anti-iced by high pressure bleed air. The system is controlled by the WING anti-ice switches on the center instrument panel (Figure 10-18).

When the switches are placed in the on position, the respective wing anti-ice solenoid valve deenergizes open, which allows engine bleed air to enter the wing leading edge (Figure 10-18). Each engine normally supplies high pressure bleed air to its respective wing; however, the bleed lines from each engine are



CITATION 650 SERIES PILOT TRAINING MANUAL

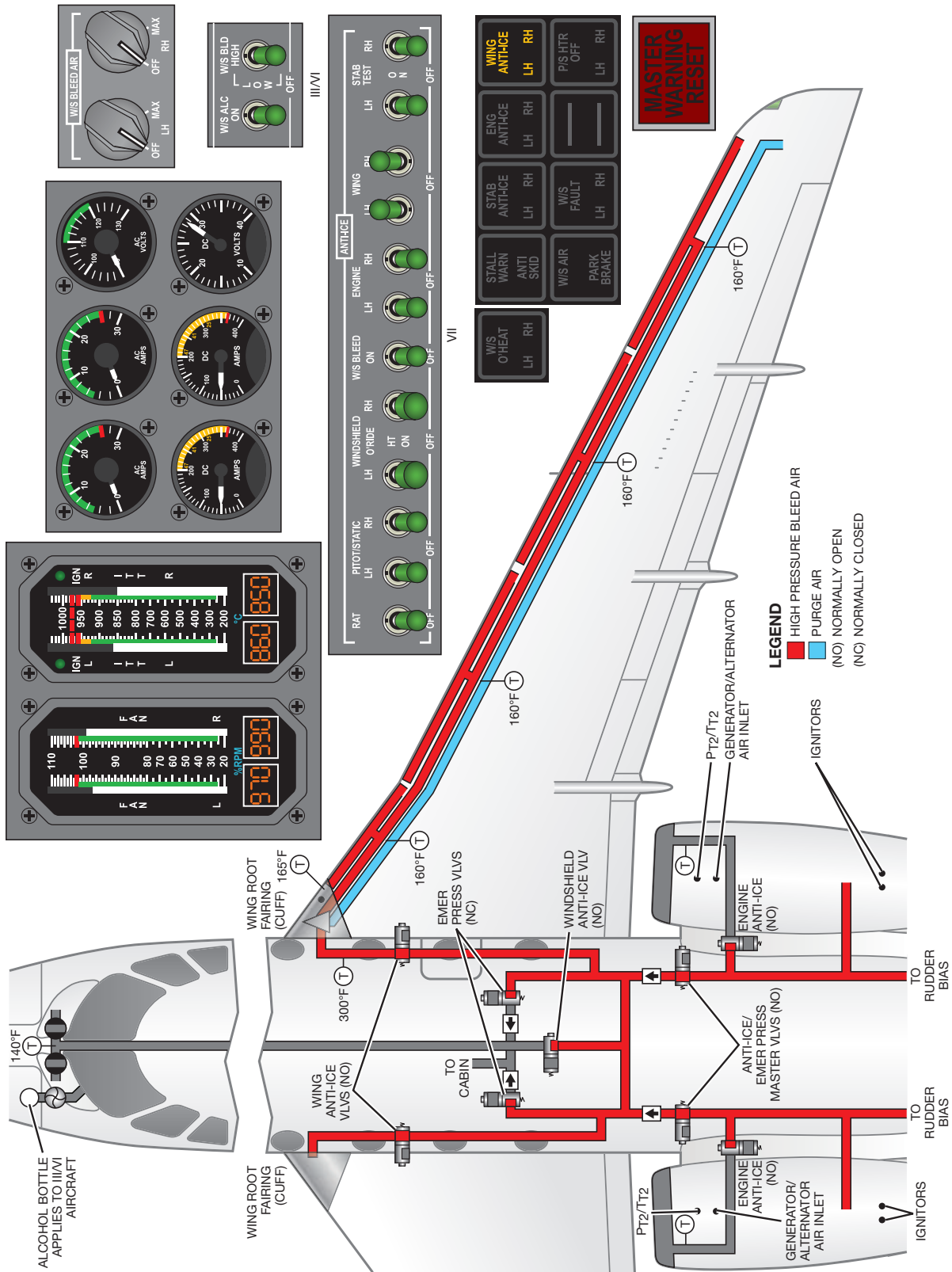
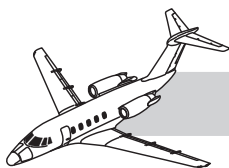


Figure 10-18. Wing Anti-Ice



interconnected to allow one engine to supply both wings if necessary provided that both wing anti-ice switches are in the ON position.

The bleed air is routed under the upper wing fairing to the leading edge where it feeds four manifold distribution (piccolo) tubes (Figure 10-19). These tubes exhaust bleed air into a narrow cavity formed by the leading edge skin and an insulated liner. The bleed air is exhausted overboard through a port on the bottom of the wing at the outboard end of each heated section.

A heat shield just forward of the fuel closure, protects the wing fuel tank from high temper-

atures. A purge air inlet in the wing root induces ram-air flow behind the heat shield to vent and to lower the temperature at the fuel closure (Figure 10-19). This air is vented to the atmosphere through the louver on the underside of the wing near the tip.

Each wing has four overtemperature switches in parallel, which are normally open. If the temperature reaches overheat conditions, the overtemperature switch closes, signalling the wing anti-ice solenoid valve to close. When the wing temperature cools down, the overtemperature switch opens, signalling the anti-ice solenoid valve to open, again furnishing bleed air to the wing.

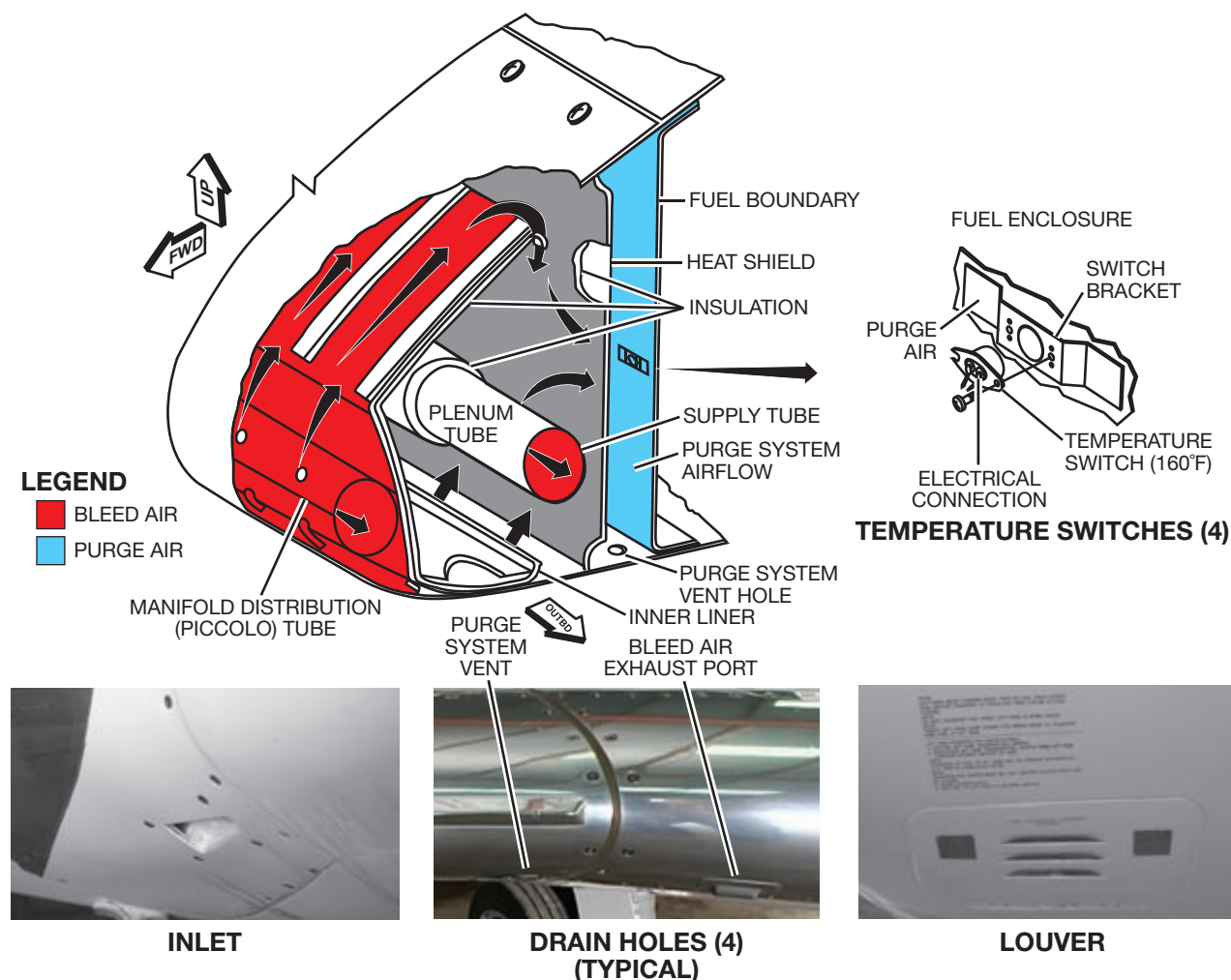


Figure 10-19. Purge Air Components



The anti-ice fail temperature switch is normally an open switch. If the bleed air temperature is below preset limits, the switch closes, illuminating the WING ANTI-ICE LH–RH annunciator and alerting the crew that bleed air to the wing anti-ice system is too cool.

NOTE

A cycling WING ANTI-ICE LH–RH annunciator indicates an overheat condition exists. A steady WING ANTI-ICE LH–RH annunciator indicates that the air temperature to the wing may be inadequate for anti-icing purposes. A steady light occurs when the system is initially turned on.

HORIZONTAL STABILIZER ANTI-ICING

Each horizontal stabilizer leading edge is anti-iced by an electrically heated parting strip and four sequentially heated shedding strips. The system operates continuously when the STAB LH–RH switch is in the ON position (Figure 10-20).

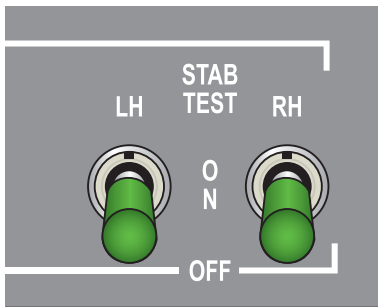


Figure 10-20. STAB Switches

Power for the system is furnished by a variable frequency alternator on each engine. Each alternator provides 115 VAC, 200–400 Hz power to the horizontal stabilizer on its respective side of the aircraft. Operation of the alternators is monitored by an ammeter for each alternator and a voltmeter. The voltmeter is selected to monitor either the left or right alternator by the two-position LH ALT–RH ALT switch on the copilot meter panel (Figure

10-21). Voltage cannot be observed until the respective STAB switch is selected to ON, which energizes the alternator field.

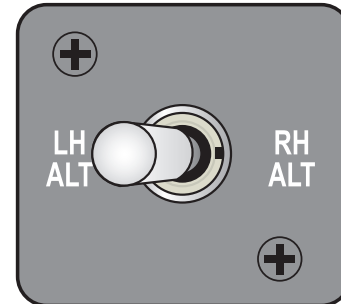


Figure 10-21. LH ALT–RH ALT Switch

Each horizontal stabilizer has a temperature sensor where the highest temperatures are reached on the parting strip (Figure 10-22). The sensor provides a temperature signal to the temperature controller, which maintains a parting strip temperature of 130–150°F. The shedding strip heating elements are powered in the following sequence:

1. Upper inboard
2. Lower inboard
3. Upper outboard
4. Lower outboard

The shedding strip heating sequence continues as long as the STAB LH–RH anti-ice switch is on. Proper operation of the system in flight is verified by the AC ammeters cycling and the STAB ANTI-ICE LH–RH annunciator remaining out.

An overheat switch on the heating blanket senses excessive temperature and protects the blanket. The switch opens at 170°F, interrupts power to the system, and illuminates the STAB ANTI-ICE LH–RH annunciator. The switch closes again at 130°F restoring power and extinguishing the STAB ANTI-ICE LH–RH annunciator.

A low temperature thermoswitch is on the parting strip to monitor minimum temperature, which when reached, illuminates the STAB

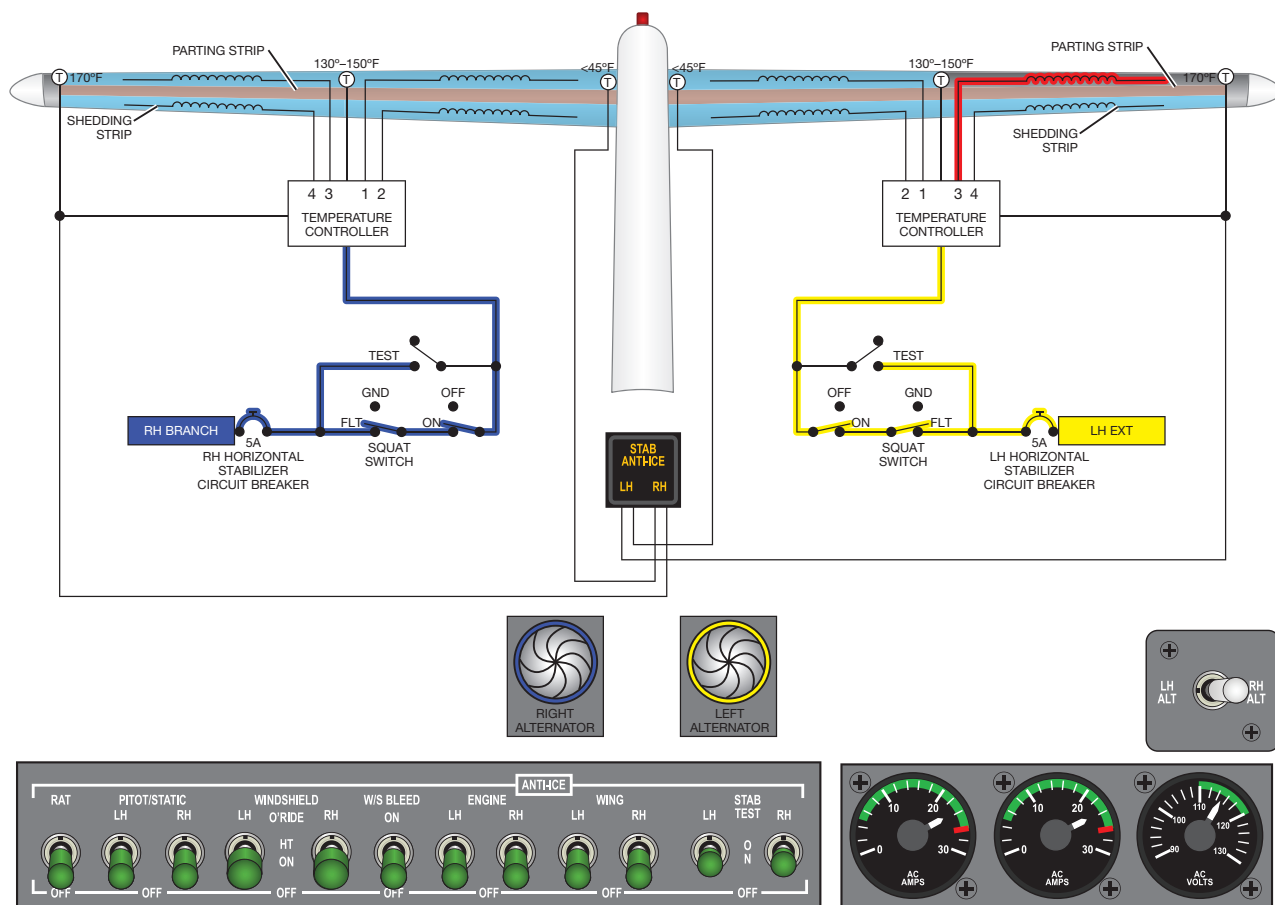
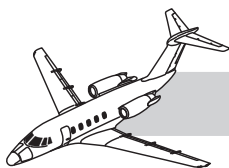


Figure 10-22. Horizontal Stabilizer Anti-Ice

ANTI-ICE LH–RH annunciator, warning the crew that the system is inoperative. The system allows an initial 20-second warm-up period before the STAB ANTI-ICE LH–RH annunciator illuminates for lower temperature. The system is inhibited during ground operation (except for testing) by the main landing gear squat switches.

System Test

The horizontal stabilizer anti-ice system is tested by placing the STAB LH–RH switches to the TEST position, then ON. The STAB ANTI-ICE LH–RH annunciator illuminates for approximately 5 seconds and then extinguishes. Check the ammeters for appropriate cycling of amps. After testing the system, turn off the STAB LH–RH switches.

NOTE

An excessive number of test cycles on the ground can damage the stabilizer leading edges.

HEATED DRAINS

The refreshment center and/or relief tubes are equipped with a heated drain which operates on DC voltage. The heated drains may be forward, midship, aft, or a combination thereof, depending on the cabin configuration. Electrical power is taken from an aft J-box electrical circuit. Power is supplied to the heated drains when normal DC power is applied to the aircraft, and the drain heater circuit breaker (on the aft J-box) is closed.



LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

EMERGENCY/ ABNORMAL

For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.

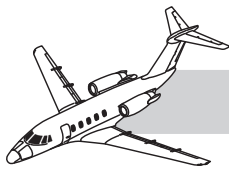


QUESTIONS

1. The incorrect statement regarding the P/S HTR OFF LH–RH lights is:
 - A. A light illuminates if power is lost to the respective angle-of-attack probe
 - B. The light illuminates if the switch is off
 - C. Illumination of the light could mean loss of power to a pitot tube heater
 - D. Illumination of the light could mean loss of power to static port heater
2. Regarding the windshield bleed air system:
 - A. The W/S BLD switch controls volume
 - B. The W/S BLD switch turns the system on only
 - C. Electrical power is necessary to open the solenoid control valve in the tail
 - D. Temperature is controlled by the manual valves
3. Regarding the use of the windshield anti-ice system:
 - A. O’RIDE should be used whenever the system is turned ON
 - B. ON should only be used in sudden severe icing conditions
 - C. O’RIDE provides a ramp-up feature for gradually heating a cold soaked windshield
 - D. O’RIDE position provides maximum windshield heat for sudden windshield icing
4. The W/S AIR light illuminates when the switch is off because:
 - A. 5 psi pressure has built up in the duct
 - B. This is a malfunction; the light is only activated by temperature
 - C. Pressure has built up in the duct and the pilot must reduce it by opening the manual valve
 - D. Air temperature in the duct is excessive
5. The rain removal system consists of:
 - A. Deploying the rain doors
 - B. Deploying the rain doors and using alcohol
 - C. Deploying the rain doors and using windshield bleed air
 - D. Deploying the rain doors and placing the W/S BLEED switch OFF
6. Regarding the windshield anti-icing system:
 - A. The left alternator completely anti-ices both windshields (pilot and copilot)
 - B. The left alternator anti-ices 2/3 of the left windshield, 1/3 of the right windshield, and the right side window
 - C. The right alternator provides full anti-icing to both pilot side windows
 - D. The windshield anti-ice system is not to be used for defogging purposes
7. Turning the engine anti-ice switches on in flight:
 - A. Activates engine ignition
 - B. Decreases ITT due to bleed air extraction
 - C. Applies 28 VDC to open the nacelle bleed air valve
 - D. Applies bleed air to the conical spinner
8. The anti-ice system must be turned on in flight when:
 - A. RAT is $< +10^{\circ}\text{C}$
 - B. OAT is $< +10^{\circ}\text{C}$ with visible moisture present
 - C. RAT is $< +10^{\circ}\text{C}$ with visible moisture present
 - D. In visible moisture with a 4° temperature dew point spread



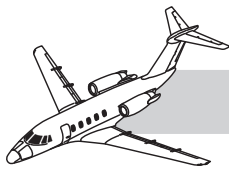
9. The W/S BLEED switch is positioned to ON and the W/S O'HEAT LH–RH annunciator illuminates and extinguishes. This indicates:
- A. Normal operation. The light was caused by initial low temperature
 - B. High temperature. The system is now inoperative
 - C. 5 psi pressure in the duct. The system is not operating normally
 - D. The system is cycling due to over-temperature
10. The engine anti-ice system includes:
- A. Electrically heated wing root fairing and lip of the engine nacelle
 - B. Electrically heated wing root fairing and generator air inlet
 - C. Bleed air heated nacelle lip and inlet temperature/pressure probe
 - D. Electrically heated inlet temperature/pressure probe and the bleed air heated nacelle lip and stator vanes
11. The wing anti-ice system:
- A. Is activated with an ice buildup of 1/4 inch
 - B. Uses power from the AC alternators to heat the leading edge
 - C. Cycles off in case of an overtemperature
 - D. Utilizes high temperature, low pressure bleed air
12. The horizontal stabilizer anti-ice system uses AC power:
- A. From the static inverters
 - B. From the alternators in parallel
 - C. To continuously heat all heating elements
 - D. From the alternator on the same side i.e., left to left and right to right



CHAPTER 11 AIR CONDITIONING

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CHAPTER 11

AIR CONDITIONING



INTRODUCTION

The Citation 650 series air-conditioning system provides conditioned air to the cabin and cockpit. The system uses engine bleed air that is processed by two pneumatic air conditioners (PACs). Temperature control normally is automatic, but a manual backup is provided. A windshield defog fan is furnished for the cockpit and in the Citation III/VI a cockpit auxiliary heater is available to supply additional warm air.

GENERAL

The aircraft is provided with an air-conditioned environment using individual temperature controls for the cabin and cockpit.

Controls on the environmental control panel are used to select the air source, control the PACs, select cabin and cockpit temperatures, and choose temperature readouts.

Air distribution uses high or low pressure bleed air from each engine and routes the air to the PACs in the tailcone compartment. The air is cooled and distributed through ducts to the cockpit and cabin outlets.



ENVIRONMENTAL CONTROL SYSTEM

DESCRIPTION

Two pneumatic air conditioners, or PACs, are in the tailcone compartment (Figure 11-1). One is for the cabin and one is for the cockpit. Normally, the right engine supplies bleed air to the cabin and the left engine supplies bleed air to the cockpit.

The PACs use high or low pressure bleed air as selected by the PAC BLD SELECT switch. An isolation valve, when open, allows both engines to supply bleed air to one PAC or one engine to supply bleed air to both PACs. The isolation valve also allows the optional auxiliary power unit (APU) to supply bleed air to both PACs (SNs 0001—0104). All other APUs provide air directly to the PACs.

Each PAC contains heat exchangers, an air cycle machine, a water separator, valves, and sensors. Bleed air entering the PAC first passes through a pressure regulating valve, which controls air volume. PAC output air is mixed with hot bleed air to produce conditioned air.

Bleed air flows through the primary heat exchanger, where it is cooled. The cooled air flows through the compressor of the air cycle machine, where it is compressed and heated.

The bleed air then flows through the secondary heat exchanger, where the air temperature is lowered further.

A thermal sensor in the duct downstream of the compressor sends a signal to the PAC O'HEAT CKPT—CAB annunciator if the air temperature is too high. The thermal sensor closes the PAC control valve which shuts off the PAC when utilizing engine bleed air. On some models, if the PAC is operated by APU bleed air, the PAC may continue to operate with the PAC control valve closed.

On SNs 0001—0104, the PACs are monitored by two temperature switches. One is past the

compressor and the other is before the turbine, past the second heat exchanger. Either switch illuminates the appropriate PAC O'HEAT CKPT/CAB annunciator and shuts down the PAC.

On SNs 0105 and subsequent, APU bleed to the PAC enters downstream from the PAC control valve. If, a PAC overheat condition occurs, the APU bleed air switch must be positioned to OFF to stop the bleed air flow to the PAC.

Ambient air enters the tailcone through the dorsal fin inlet (Figure 11-2). A fan on the turbine shaft blows the air across the PAC heat exchangers. The air is then exhausted overboard through louvers above the right engine pylon (Figure 11-3).



Figure 11-2. Dorsal Fin Inlet



Figure 11-3. PAC Heat Exchanger Exhaust

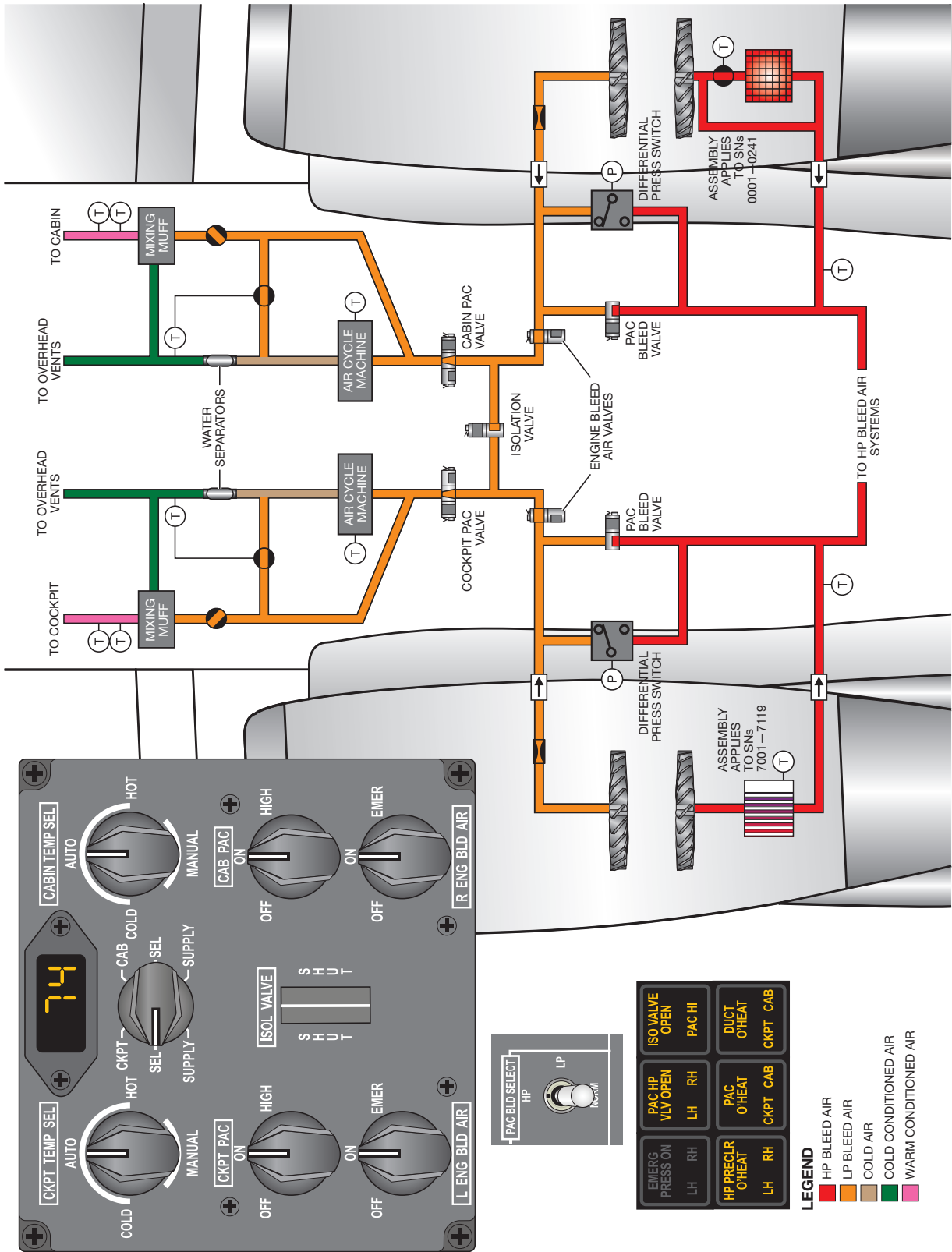


Figure 11-1. Air Conditioning System Diagram



On SNs 0001—0104 the turbine shaft in the PAC has no fan for drawing air across the heat exchangers. A duct connects the dorsal fin inlet directly to the heat exchangers and to an overboard vent. RAM-air flow is provided in-flight, but not on the ground. A hydraulically operated ECU fan draws air through the duct and across the heat exchangers during ground operation.

The fan does not operate in-flight. The fan operates only on the ground when main hydraulic system pressure is available and when at least one PAC switch is selected to ON or HIGH.

The air leaving the secondary heat exchanger passes through the cooling turbine, where the air temperature and pressure are substantially reduced by expansion and extraction of heat. This energy is used to drive the compressor via the turbine.

The cold air then enters the water separator, where excess moisture is removed. Water particles entering the separator are coalesced until drops form. The air then passes through a swirl chamber, where the water drops are centrifuged out of the air.

The collected water is sprayed on the secondary heat exchanger for greater cooling efficiency. If normal airflow through the coalescer is restricted, a bypass valve opens, allowing air to flow around it. Air leaving the water separator passes a temperature sensor, which controls an anti-icing valve.

The anti-icing valve mixes hot bleed air with the cool turbine exhaust. The mixing action maintains air temperature at greater than 35°F (1.6°C) and prevents freezing of water in the water separator.

The cold air flows from the PACS to the distribution system. Air temperature to the overhead ducts is not further modified. Air to the foot warmers, cabin armrest warmers, cockpit armrest outlet, and windshield mixes with hot bleed air for temperature control. The temperature control valve mixes hot bleed air with

cold conditioned air, and is controlled either automatically or manually.

CONTROLS AND INDICATIONS

The environmental control panel has selectors with AUTO and MANUAL positions for controlling cockpit and cabin temperature. The selected temperature is adjusted within a range of 65°F–85°F (18°C–29°C) in the AUTO position. The temperature control valves automatically modulate to maintain the selected temperature (Figure 11-1).

A digital readout and a selector for readout control is between the temperature selectors. The positions are CKPT, SEL, and SUPPLY, and CAB, SEL, and SUPPLY.

In the CKPT or CAB positions, the current cockpit or cabin temperature is displayed.

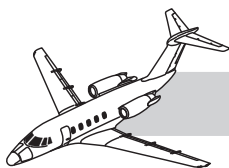
In the SEL position, the specific cockpit or cabin temperature is set using the CKPT or CAB TEMP SELECT knob when in the auto mode.

In the SUPPLY position the temperature of the associated cabin or cockpit tailcone duct is displayed.

When the CKPT or CAB TEMP SEL knob is selected to MANUAL, the cabin, cockpit, and supply duct temperatures are still monitored, but if the display is switched to SEL, the readout is meaningless.

OPERATION

If the automatic temperature control malfunctions, rotate the temperature selector to the MANUAL position. Doing so rotates the control to a detent, causing it to stay centered at the 6 o'clock position. The control then acts as a rotary toggle switch, and the temperature control valves can be moved directly toward the warmer or cooler position.



Holding the control toward COLD rotates the valve toward the closed position; releasing the control stops the valve rotation.

Holding the control toward HOT rotates the valve toward the open position; releasing the control stops the valve rotation.

Moving the valves from fully open to fully closed requires approximately 25 seconds. If electrical failure occurs, then temperature control is lost, and the temperature control valves remain in the last selected position, regardless of temperature selector position (AUTO or MANUAL).

AIR DISTRIBUTION

CABIN DISTRIBUTION

The cabin PAC supplies the cabin air distribution system with conditioned air. Emergency pressurization air is supplied from either or both engines. Fresh air is supplied from the ram-air ducts and is available only if the cabin is unpressurized.

Cabin air is routed forward to the sidewall armrest and foot warmers. Left cabin air is routed only to the aft side of the main entrance door.

COCKPIT AIR DISTRIBUTION

The cockpit is supplied conditioned air or fresh air from the ram-air ducting. If the cockpit PAC is not operating, a check valve in the cockpit duct opens, allowing conditioned air from the cabin to be routed to the armrest outlets, side windows, foot warmers, and windshield defog nozzles.

Defog airflow can be boosted by a two-speed blower. The blower is controlled by the W/S DEFOG switch on the copilot side lower instrument panel. The switch has LO–OFF–HI positions.

Air flow control knobs on those aircraft equipped with them are directly forward of

each CB panel and control airflow through the foot warmers and side window vents. Another vent knob on the copilot side lower instrument panel controls airflow inside the windshield.

OVERHEAD DISTRIBUTION

Overhead cold air starts at the PAC assembly. The overhead ducts continue forward to terminate in the cockpit. Fresh air is available through the ram-air ducts only if the cabin is unpressurized.

WEMAC BOOST SYSTEM

The WEMAC boost system increases airflow through the overhead ducts via an electric blower in the aft vanity closet.

The squirrel cage-type blower connects to a crossover duct, which connects the left and right overhead ducts. WEMAC boost also eliminates fogging from the WEMAC outlets by mixing drier cabin air with the overhead cold air. A check valve on the blower air inlet prevents reverse flow when the blower is not operating.

The WEMAC boost switch is on the copilot side instrument panel. The WEMAC boost blower is protected by a circuit breaker in the tailcone J-box.

OPTIONAL FLOOD COOLING

The flood cooling system supplies a high volume of cool air directly from the PACs through an air outlet grill in the upper portion of the aft cabin (Figure 11-4). The system diverts most of the cool air from the overhead WEMAC outlets and is intended for cooling purposes only. Flood cooling is controlled by a third position on the WEMAC BOOST switch.

The OFF position directs the overhead cool air through the normal overhead distribution system. The FLOOD COOLING position opens a door in the crossover plenum, thereby directing cool air through the flood cooling outlet directly into the aft cabin. When the flood cool-

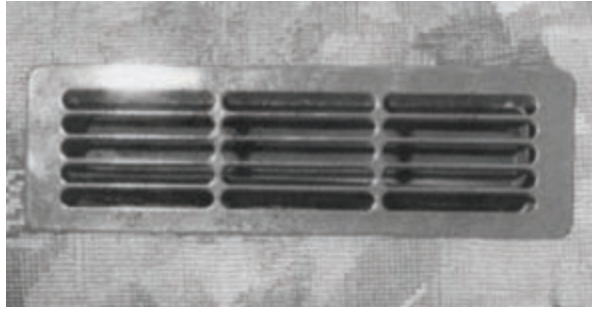


Figure 11-4. Flood Cooling Outlet

ing door is open, airflow is partially blocked to the overhead WEMAC outlets. Some air flows to the overhead cabin and cockpit outlets when the flood cooling system is operating.

The flood cooling plenum is across the upper forward side of the aft pressure bulkhead connecting the left and right overhead ducts.

The FLOOD COOLING position also powers the WEMAC boost blower. The blower, which is connected to the flood cooling crossover plenum, increases airflow from the flood cooling outlet.

Placing the WEMAC BOOST switch in the upper position powers the WEMAC BOOST blower, but does not open the flood cooling door, and allows for normal WEMAC boost as previously discussed.

NOTE

Aircraft equipped with APUs having the HIGH position associated with the APU bleed air switch must have the optional flood cooling system incorporated.

FREON AIR CONDITIONING SYSTEM

The freon air-conditioning system, if installed, provides supplemental cooling and circulation on the ground and at low altitude. The system can be operated either independently or in conjunction with the standard environmental

system. If the freon system is installed, the flood cooling option is also incorporated.

The freon air-conditioning system consists of two evaporator/fans, one forward and one aft. The forward evaporator/fan is in the aisle floor, aft of the cockpit, and uses a squirrel cage fan for airflow distribution. A grille is in the floor for an air outlet. Adjustable louvers directly below the grille direct airflow toward the cockpit or cabin.

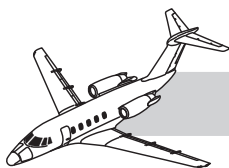
The aft evaporator/fan is similar to the forward unit and is in the tailcone directly behind the aft pressure bulkhead. The aft fan blows air through the cabin overhead WEMAC vents toward the cockpit. The WEMAC fan can also be used to increase airflow through the overhead ducts.

The flood cooling system provides airflow assistance. Using the flood cooling system with freon air-conditioning opens the flood cooling door, allowing airflow through the flood cooling outlet on the upper portion of the aft pressure bulkhead, directly into the aft cabin (Figure 11-5).



Figure 11-5. Air Outlet (Freon System)

The flood cooling switch ON position shuts off airflow through the overhead cabin and cockpit ducts, opens the flood cooling outlet door, and powers the WEMAC boost fan. Using both the freon and flood cooling systems max-



imizes airflow into the cabin. Using only the freon air-conditioning system and the WEMAC boost fan, with flood cooling OFF, directs the forward evaporator/fan louvers toward the cockpit, maximizing airflow into the cockpit.

The electrically driven freon compressor and receiver/dryer are in the tailcone compartment (Figure 11-6). The freon air conditioner is controlled from the panel directly forward of the copilot CB panel (Figure 11-7). The freon system can be used on the ground and up to an altitude of 18,000 feet. Only the fans can be used at any altitude.

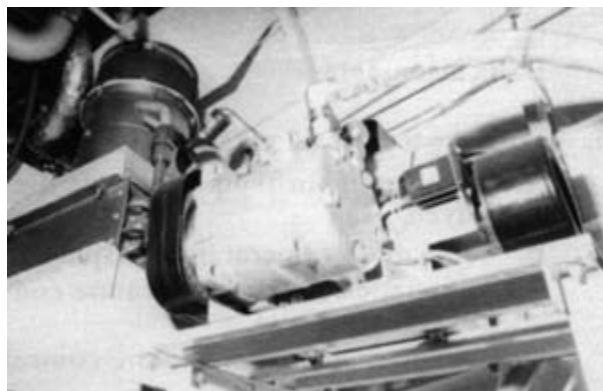


Figure 11-6. Compressor and Receiver Dryer (Freon System)



Figure 11-7. Control Panel (Freon System)

The freon air conditioner master switch (A/C OFF–FAN–ON) controls power to the system. The ON position activates the compressor and the evaporator fans. The compressor is powered by an electric motor. The fans operate at various speeds, depending on the position of their respective switches (LOW–MED–HI). The FAN position activates the evaporator/fans only; the compressor remains off.

The A/C TEMP switch is a thermostatic control, which cycles the compressor when the selected temperature is reached. The temperature sensor is on the cabin side of the forward evaporator.

A binary pressure switch in the freon air conditioner shuts off compressor operation if system pressure is too low, which can be caused by excessively cold ambient temperatures. The compressor also shuts down under conditions of excessively high pressure (overheating). Do not operate the freon air conditioner at ambient air temperatures below -20°C (-4°F).

When only one generator is operating, the freon air conditioner must be OFF because of high current demands by the air conditioner.

The freon air-conditioning system adds a total of 139.4 pounds to the aircraft basic empty weight.

AUXILIARY HEATER

An auxiliary heater in the cockpit ducts provides additional volume and heat for the cockpit and windshield defog systems in the Citation III and VI.

The auxiliary heater consists of a 28 VDC blower motor and three electrical heating elements in a metal housing. The heater is in the pressurized cabin at the aft pressure bulkhead and connects to the right side cockpit air ducts. The blower motor draws in cabin air and passes it over the electrical heating elements into the cockpit distribution ducts.

Two switches on the copilot instrument panel are used to control the auxiliary heater. The CKPT AUX FAN switch operates the blower



only. The CKPT AUX HEATER switch operates the fan and the heater elements. Both switches have LOW–OFF–HI positions.

To increase or decrease blower speed, select LOW or HI respectively on the CKPT AUX FAN switch.

The LOW position on the CKPT AUX HEATER switch energizes two of the three heating elements and operates the blower at low speed. The HI position energizes all three heater elements and operates the blower at high speed.

An airflow sensor near the blower inlet prevents heater elements from energizing without blower operation. If the sensor detects airflow, it closes to complete the heater element circuit.

If air temperature exceeds set limits it removes power from the heater elements, then an overheat sensor sends a signal to the AUX HTR O'HEAT annunciator. An additional cool-down sensor maintains blower operation after heater use until the air temperature drops below 130°F (54°C).

NOTE

If an overheat condition occurs, the cockpit auxiliary heater shuts down and the AUX HTR O'HEAT annunciator illuminates. If the heater is turned off automatically or manually while still hot, the fan runs until the heater cools, provided electrical power is available. If the battery is the only power source, do not turn the battery switch off while the fan is running.

Circuit breakers and relays for the auxiliary heater system are in the auxiliary power J-box on the upper left side of the tailcone.

BAGGAGE HEATER

An electrically operated heater is in the baggage compartment forward of the access door (standard in Citation III/VI; optional in Citation VII). Baggage compartment heat is con-

trolled with the BAG HTR ON–OFF switch. The ON position energizes the resistance elements and a blower with DC power. The blower circulates air across the resistance elements to increase the air temperature in the baggage compartment.

The circuit is monitored by ambient temperature switches, which control the temperature in the baggage compartment between 70°F and 90°F (21°C and 32°C). An overtemperature switch protects the heater against overheating. If output temperature exceeds 180°F (82°C), the BAG HTR O'HEAT annunciator illuminates and the heater automatically shuts down. The *AFM* calls for the switch to be placed in OFF under such conditions.

BAGGAGE SMOKE DETECTOR

An optional smoke detector system detects smoke in the baggage compartment. The system consists of a sensor in the baggage compartment and a red BAGGAGE SMOKE annunciator in the cockpit. To test the system, press the BAGGAGE SMOKE annunciator and then verify illumination of the BAGGAGE SMOKE annunciator and the MASTER WARNING RESET switchlights.

LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

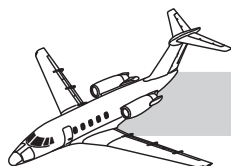
EMERGENCY/ ABNORMAL

For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.



QUESTIONS

1. The cockpit PAC:
 - A. Normally receives engine bleed air from the left engine
 - B. Normally receives engine bleed air from the right engine
 - C. Furnishes conditioned air for the cockpit and cabin foot warmer outlets
 - D. Furnishes conditioned air for the overhead ducts
2. Air from the overhead outlets comes from the:
 - A. Cockpit PAC only
 - B. One or both PACs
 - C. Cabin PAC only
 - D. None of the above
3. Air in the overhead outlets:
 - A. Varies in temperature between 65°C and 85°C
 - B. Can be heated by turning the defog fan on
 - C. Normally is a warm temperature
 - D. Normally is a cool temperature
4. If a cockpit PAC fails:
 - A. The cabin PAC furnishes air to the LEFT overhead outlets only
 - B. Conditioned cabin air is routed through cockpit ducts
 - C. Emergency pressurization must be selected
 - D. Emergency pressurization is automatically selected by the system
5. When the cabin temperature SEL position is selected and the cabin temperature control is placed in MAN, the temperature display:
 - A. Reads cabin temperature
 - B. Reads selected temperature
 - C. Reads 888
 - D. Is meaningless
6. With the temperature knob in CAB, the:
 - A. Existing cabin temperature is displayed
 - B. Cabin duct temperature is displayed
 - C. Automatic cockpit temperature control is inoperative
 - D. Cockpit manual temperature control is inoperative
7. Environmental system control allows:
 - A. Both PACs to receive air from one engine
 - B. One PAC to receive air from both engines
 - C. Both PACs to receive either high or low pressure bleed air
 - D. All of the above
8. The PAC selector switch:
 - A. When positioned to HIGH, allows a greater flow of bleed air into the PAC
 - B. When positioned to HIGH, allows only high pressure bleed air into the PAC
 - C. When positioned to OFF, stops the bleed air from the associated engine from entering the aircraft
 - D. Must be positioned to OFF prior to engine start
9. The source of bleed air for cabin pressurization with the L ENG BLD AIR switch in EMER and the R ENG BLD AIR switch in ON is:
 - A. Both engines
 - B. Left engine only
 - C. Right engine only
 - D. Ram air
10. In order for the PACs to receive high pressure bleed air, the PAC BLD SELECT switch must be positioned to:
 - A. HP
 - B. LP
 - C. NORM
 - D. Either A or C (taxiing)

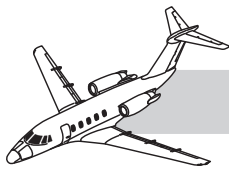


CHAPTER 12

PRESSURIZATION

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CHAPTER 12

PRESSURIZATION



INTRODUCTION

The pressurization system on the Citation 650 series aircraft maintains a lower cabin altitude than actual aircraft altitude. The system distributes engine bleed air, conditioned through the pneumatic air conditioning (PAC) units, into the cabin and cockpit. The degree of pressurization (cabin altitude) is determined by the amount of air permitted to escape through the outflow valves. The system can hold cabin altitude to 8,000 feet when the aircraft is at an altitude of 51,000 feet.

GENERAL

Cabin pressurization requires a constant air source and a means of controlling air flow into and out of the aircraft to achieve the desired differential pressure and cabin altitude. Air inflow is maintained at a constant rate through a wide range of engine power settings. Air

outflow is controlled by two outflow valves on the aft pressure bulkhead. The air-conditioning system provides the inflow air. The position of the outflow valves is normally controlled by the cabin pressurization controller or by a manual control system.



COMPONENTS

The pressurization control system consists of a manual control valve and selector on the center pedestal, a remote digital controller and two outflow valves on the aft pressure bulkhead, interconnected at the control chambers.

The controller provides two modes of automatic control: ALTITUDE SELECT and AUTO SCHED. A manual control switch (MAN/NORM) is available for manual control of pressurization (Figure 12-1).

Both outflow valves have altitude limit control valves to hold cabin altitude at approximately 13,500 feet. Each outflow valve also has a pneumatic cabin-to-atmosphere differential pressure limiter, which allows the outflow valves to relieve any pressure in excess of 9.5 psid.

Primary Outflow Valve

The primary outflow valve contains an electropneumatic transfer valve (torque motor), positive differential pressure control and an altitude limiter control (Figure 12-1).

The torque motor moves a flapper valve on demand by the controller. The flapper valve movement applies vacuum to the outflow valve control chamber, adjusting cabin pressure up or down. The secondary outflow valve operates in conjunction with the primary outflow valve because of the interconnection between the control chambers.

Secondary Outflow Valve

The secondary outflow valve contains an isobaric hold chamber and isolation valve instead of a torque motor (Figure 12-1). It also has a positive differential control and an altitude limit control, which operate together the same as the primary valve. The right outflow valve static port provides sensing for the secondary outflow valve differential pressure limiter.

The isolation valve is normally powered open by the controller, making the hold chamber ineffective. When the isolation valve fails closed, cabin altitude is controlled by the manual control.

If the primary valve fails because of electrical or aircraft vacuum failure, then the secondary valve remains manually controllable.

Pressurization Controller

The pressurization system uses an electronic signal from the pressurization controller to drive a torque motor in the primary outflow valve (Figure 12-1). The torque motor opens a valve, allowing vacuum into the outflow valve control chamber, thereby opening the outflow valve and decreasing cabin pressure.

When the signal is decreased, the vacuum is reduced, increasing the pressure in the control chamber. The outflow valve modulates toward closed and increases cabin pressure. The controller determines the signal after sensing cabin and outside pressure and then comparing the signal to the pressurization schedule.

CONTROLS AND INDICATIONS

PRESSURIZATION CONTROL PANEL

The MAN–NORM switch determines whether the aircraft pressurization is controlled manually or automatically (Figure 12-1).

The MAN position disables the cabin pressure selector and enables the manual control toggle and RATE knob, which are used to manually control the cabin altitude and rate of change.

The NORM position enables automatic pressurization control, AUTO SCHED or ALTITUDE SELECT.





AUTO SCHED determines cabin altitude based on a preprogrammed automatic schedule for aircraft altitude and pressurization (Figure 12-2).

ALTITUDE SELECT pressurizes the aircraft according to the settings on the cabin pressure selector (Figure 12-1). After takeoff the cabin pressurizes to the selected altitude at the selected rate.

The CAB ALT WARN MUTE/TEST button near the CABIN DUMP switch silences the aural warning horn (Figure 12-1). On the ground the MUTE/TEST button is used to test the cabin altitude warning horn.

The FAULT annunciator on the cabin altitude selector illuminates briefly during the self test.

When the squat switches are in the air mode the test function is disabled. If cabin altitude rises above 10,000 feet, then an aural warning horn sounds in the cabin. Pressing the MUTE/TEST button silences the horn.

In the AUTO SCHED and ALTITUDE SELECT modes, the controller limits maximum differential pressure to 9.3 psid. In AUTO SCHED mode, maximum differential pressure is reached at 37,000 feet. While in ALTITUDE SELECT mode, it is reached at approximately

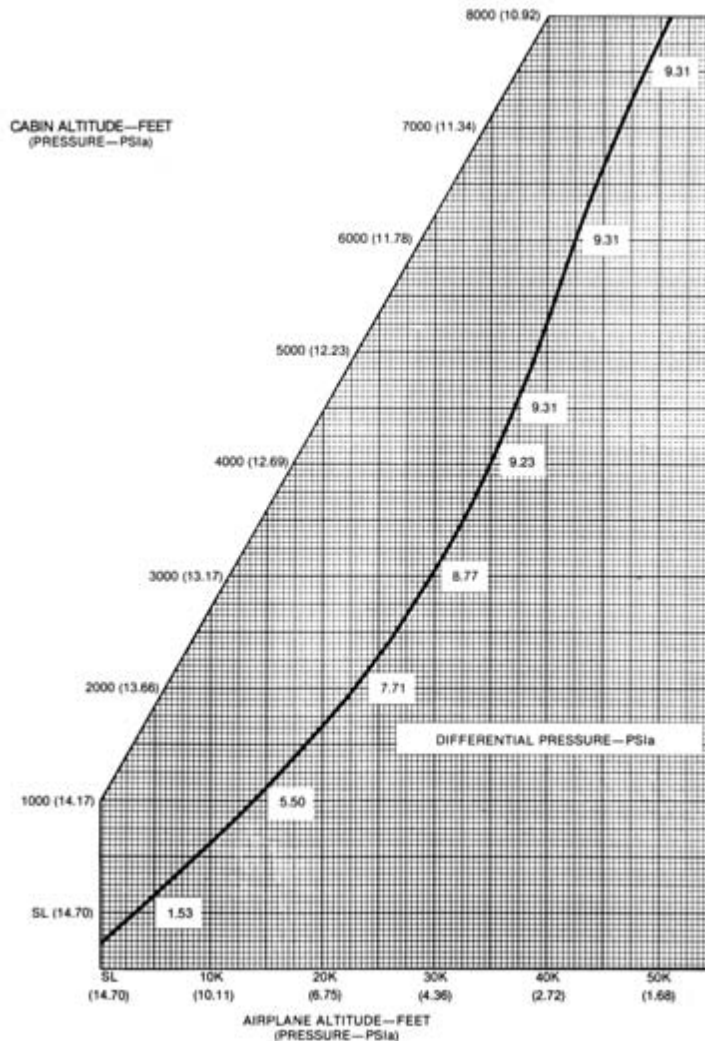
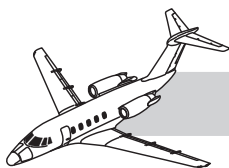


Figure 12-2. Automatic Schedule Graph



26,000 feet. Upon reaching maximum differential pressure, the controller climbs the cabin as required to maintain a 9.3 psi differential.

CABIN ALTITUDE SELECTOR

During NORM operation, the selector inputs to the controller are:

- Departure/landing field elevation or cruise altitude (knob A)
- Barometric pressure (knob B)
- Rate of climb or descent (knob R)

All settings are made before takeoff, with barometric pressure set at the departure field, and then reset for the destination field prior to descent. The knobs can be adjusted at any time.

The rate of change of cabin altitude is adjustable from 150–2,500 fpm rate of climb and 90 to 1,500 fpm rate of descent. Intermediate settings give proportional rates. The recommended “pip” selection gives a 500 fpm rate of climb and 300 fpm rate of descent.

SYSTEM FAULT ANNUNCIATOR

The pressurization system FAULT annunciator is on the cabin altitude selector between knob R and B (Figure 12-1). When power is applied, the system initiates a self-test. The annunciator illuminates during the test and then extinguishes, indicating satisfactory system operation. If the annunciator remains illuminated, then the self-test failed.

CABIN DUMP SWITCH

The red guarded CABIN DUMP switch allows for rapid depressurization (Figure 12-1). The CABIN DUMP switch is used to completely open both outflow valves. Because of altitude limit valves, cabin altitude may not dump above $13,500 \pm 1,500$ feet. For complete depressurization, bleed air to the CKPT and CAB PACs must be turned off, to include APU bleed air as applicable, and emergency pressurization valves reset after they open automatically.

OPERATION

AUTOMATIC CONTROL

To operate the pressurization system automatically, place the MAN–NORM switch in the NORM position, and place the AUTO SCHED–ALTITUDE SELECT switch either to AUTO SCHED or ALTITUDE SELECT (Figure 12-1).

AUTO SCHED automates the system using the automatic schedule logic and the A, B, and R knobs.

While the aircraft is on the ground with the engines operating below 70–85% N_1 , a full open signal is sent to the outflow valves. With both outflow valves open, the cabin pressurizes down approximately 80 feet. If N_1 exceeds 70–85%, the system enters the pressurization mode and the controller pressurizes the cabin down an additional 180 feet at the selected rate. Retarding the throttles below 70–85% reopens the outflow valves.

When weight is removed from the squat switch on takeoff, the system enters the flight mode. The controller examines the scheduled cabin altitude for current aircraft altitude, compares it with the knob A setting, and then selects the higher of the two.

If the selected cabin altitude is greater than the present cabin altitude the controller climbs the cabin at a rate no greater than the limiting rate. The controller repeats this process several times each second.

If the controller selects a cabin altitude below the present cabin altitude, as is the case when flying from a high altitude airport to a lower one with the destination set by knob A, the descent rate is clamped at zero to prevent the cabin from descending. This clamp is released if the aircraft descends or levels off for one minute. This feature prevents the cabin from descending and later climbing because the aircraft is climbing. During cruise and descent, the controller maintains the cabin altitude as determined by the schedule or des-



mination altitude.

Upon landing, the squat switch places the system in the landing mode. If cabin pressure is greater than outside pressure, the controller depressurizes the cabin at the limiting rate for 1 minute. After 1 minute, the system enters the ground mode and both outflow valves are fully opened.

ALTITUDE SELECT deactivates the automatic schedule logic and the down rate clamp, and also uses knobs A, B, and R. Prior to takeoff, the selector should be set to the cruise altitude, the rate marks should be aligned, and the barometric pressure entered. After takeoff, the system takes the cabin to the pressure for the selected cruise altitude at a rate of 500 fpm (on the mark).

If the selected cruise altitude is exceeded, the cabin may have to climb at a rate higher than selected to stay within differential pressure limits. When starting the descent, the landing field elevation and barometric pressure must be selected. The cabin descends to the selected field elevation at the selected rate (300 fpm on the mark).

MANUAL CONTROL

When the MAN–NORM switch is selected to MAN or if a DC component fails, then manual pressurization control is activated (Figure 12-1). The isolation valve in the secondary outflow valve assembly closes, allowing the isobaric hold chamber to temporarily maintain present cabin altitude.

To increase or decrease cabin altitude, hold the red manual control toggle (cherry picker) in the UP or DN position and adjust the rate knob to the desired rate. Manual control uses cabin air for positive pressure and outside air for vacuum. This system is independent of aircraft vacuum or electrical power.

Always start the manual RATE knob in the MIN position, then adjust it to the desired rate. Manual control must be used if main DC power or vacuum failure occurs. Manual con-

trol also must be used if the controller fails, as indicated by the FAULT annunciator.

WARNING

If cabin altitude rises above 8,500 feet, a barometric pressure switch illuminates the CABIN ALT 8500 FT annunciator. If cabin altitude rises above 10,000 feet, a dual barometric switch system activates an aural warning.

To silence the warning horn, press the CAB ALT WARN MUTE/TEST button near the dump switch (Figure 12-1). The system has dual barometric switches, audible warning units, and speakers. The two systems are wired separately to the remote audio amplifier, where the signals are mixed for redundancy in both pilot and copilot speakers and headphones.

On the ground, the CAB ALT WARN MUTE/TEST button is used to test the cabin altitude warning horns.

DEPRESSURIZATION

To depressurize the cabin activate the red guarded CABIN DUMP switch as previously described (Figure 12-1). If the system is in manual control mode, then the manual control toggle must be held in the UP position and the RATE knob adjusted to MAX.

LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

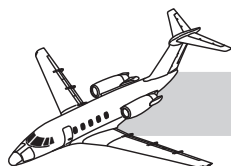
EMERGENCY/ABNORMAL

For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.



QUESTIONS

1. If the aircraft vacuum to the primary outflow valve is lost, then cabin pressure:
 - A. Goes to zero differential pressure as the aircraft depressurizes
 - B. Goes to the maximum limits allowed by the outflow valves
 - C. Stabilizes at about 13,500 feet
 - D. Remains at the present pressure
2. The emergency dump switch:
 - A. Operates with or without electrical power
 - B. Functions with or without aircraft vacuum
 - C. Is intended for ground use only
 - D. Depends on aircraft vacuum to operate
3. In AUTO SCHED position, the pressurization controller:
 - A. Pressurizes the cabin to the higher of the knob A selection or scheduled cabin altitude
 - B. Always selects maximum differential pressure
 - C. Always climbs or descends the cabin at the selected rate
 - D. Cannot exceed the maximum selected rate
4. In ALTITUDE SELECT position of the pressurization controller:
 - A. The cabin pressurizes to the higher of the knob A selection or scheduled cabin altitude
 - B. Only maximum differential pressure can be selected
 - C. The cabin climbs or descends at the selected rate
 - D. The maximum rate of climb or descent cannot be exceeded
5. In the manual pressurization control mode:
 - A. Electrical power is necessary
 - B. Aircraft vacuum is not used
 - C. Bleed air is used as a source of vacuum pressure
 - D. Maximum differential pressure can be exceeded

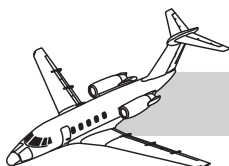


CHAPTER 13

HYDRAULIC POWER SYSTEM

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CHAPTER 13

HYDRAULIC POWER SYSTEM



INTRODUCTION

The Citation 650 series aircraft hydraulic system is normally pressurized by two engine-driven pumps, one on each engine. The system provides pressure for the wheel brakes, spoilers, aileron boost, ECU fan (III, SNs 0001-0104), landing gear, thrust reversers, and nosewheel steering. An optional auxiliary power unit (APU) driven pump can also pressurize the entire system. Some subsystems can be actuated by an electrically-driven auxiliary pump. System operation is monitored by annunciators, and pressure and volume indicators in the cockpit.

GENERAL

The aircraft uses a full-time, high pressure hydraulic system (Figure 13-1) pressurized to 3,000 psi by engine-driven hydraulic pumps on each engine. The pumps are supplied with fluid from a reservoir through electrically operated firewall shutoff valves, controlled from the cockpit.

An electric auxiliary hydraulic pump, controlled by the AUX HYD PWR switch (Figure 13-1), can be used to pressurize the roll control spoilers, wheel brakes, and spoiler hold-down system.



13 HYDRAULIC POWER SYSTEM

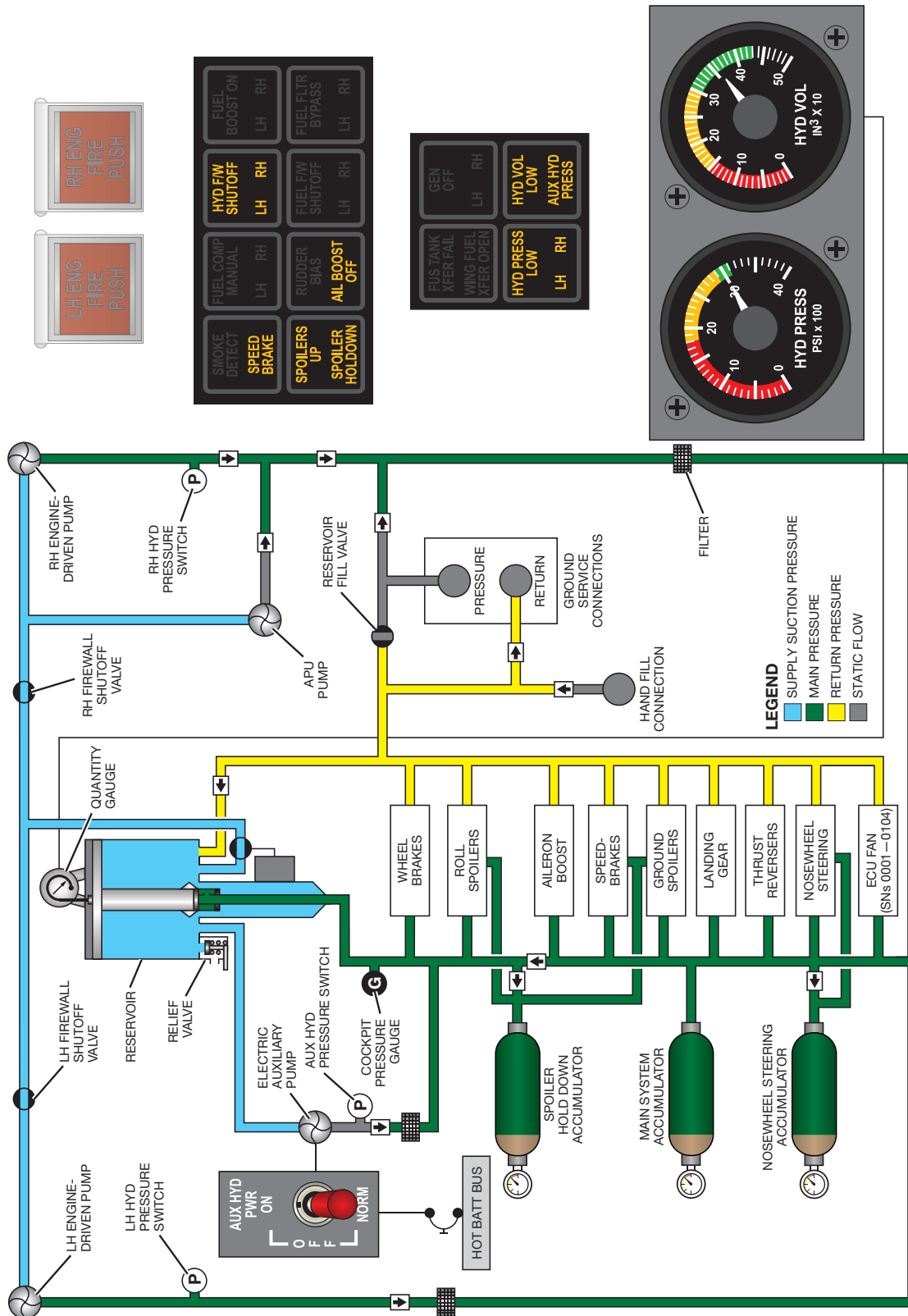


Figure 13-1. Hydraulic System Schematic



A variable displacement pump, driven by an optional onboard APU can be used to pressurize all hydraulically actuated subsystems.

A pressurized hydraulic reservoir prevents air from entering the system.

Annunciators provide indication of low reservoir fluid level, low hydraulic pressure from the engine-driven pumps, closed firewall shut-off valves, and auxiliary pump operation.

The hydraulic system uses MIL-H-83282 fire-resistant hydraulic fluid (Brayco), a synthetic, hydrocarbon-based fluid. The fluid is not caustic, has little effect on paint or rubber, and has a high lubricity rating, which extends component life.

COMPONENTS

RESERVOIR

The reservoir (Figure 13-2) is on the left side of the tailcone, above the access door. Reservoir fluid is pressurized to 50 psi by hydraulic system pressure applied to a small piston in the reservoir neck.

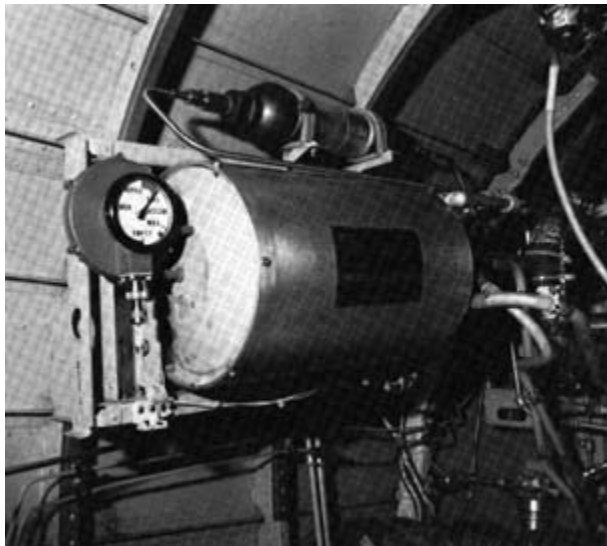


Figure 13-2. Reservoir

Equipment such as a ground hydraulic service unit (mule) or a hand-operated pump must be used to service the reservoir. Servicing connections for the mule are on the right side of the fuselage, below the right engine (Figure 13-3). Servicing connections for the hand operated pump are located in the tailcone compartment on the righthand side (Figure 13-3).

A quantity gauge on the aft end of the reservoir is mechanically actuated by the movable main piston. REFILL, FULL, and MAX indications correspond to 310-, 360-, and 500-cubic inch levels, respectively. A hash mark between FULL and MAX indicates 425 cubic inches. The gauge needle points to the hash mark when the hydraulic system is unpressurized. The needle points to FULL when the hydraulic system is pressurized. The quantity gauge provides electrical signals to the HYD VOL quantity indicator in the cockpit (Figure 13-4).



Figure 13-3. Servicing Connections

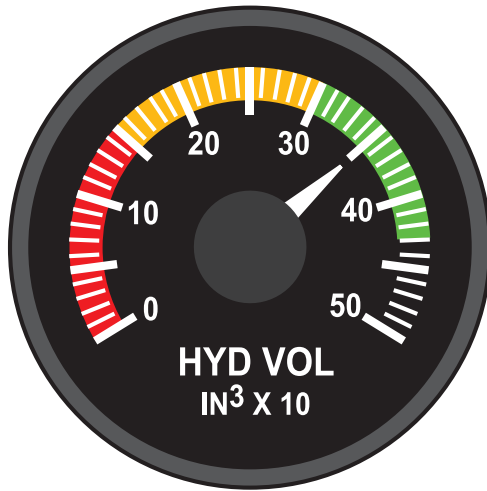


Figure 13-4. HYD VOL Gauge

If fluid level drops to 150 cubic inches a cam on the cylinder (Figure 13-5) actuates the emergency level mechanism which closes the suction line to the engine-driven pumps. This preserves fluid for emergency use by the electric auxiliary hydraulic pump. Simultaneously, the HYD VOL LOW annunciator illuminates, followed by illumination of both HYD PRESS LOW annunciators.

ENGINE-DRIVEN PUMPS

The variable displacement, engine-driven pumps in the accessory section of the engines maintain system pressure at 2,800 to 3,000 psi. The pumps operate whenever the engines are in operation. Either pump can operate all subsystems.

When the engines are started, the engine-driven hydraulic pumps draw fluid through open fire-wall shutoff valves (Figure 13-1). As pressure increases to 2,400 psi, the HYD PRESS LOW LH–RH annunciators extinguish. At 3,000 psi, with no subsystem actuated, pump output is required only to compensate for system leakage and for pump cooling. A relief valve limits pressure to 3,450 psi if a pump malfunctions. If a pump fails, the applicable HYD PRESS LOW annunciator illuminates.

If both engine-driven pumps are inoperative or low pressure is detected in the system,

pressure is supplied automatically by the auxiliary hydraulic pump to the roll spoilers and wheel brakes provided the AUX HYD PWR switch is in NORM (Figure 13-1). Operation of the auxiliary hydraulic pump is indicated by the illumination of the AUX HYD PUMP ON annunciator (SNs 0001–0155) not incorporating SB 650-29-10. SNs 0155 and subsequent and SNs 0001–0155 incorporating SB 650-29-10 have an AUX HYD PRESS annunciator that illuminates any time the auxiliary hydraulic pump is generating pressure. The auxiliary hydraulic pump, when operating, also provides holddown pressure for the speedbrakes, ground spoilers, roll spoilers, and pressure for hydraulic brake operations.

ELECTRICALLY DRIVEN AUXILIARY PUMP

The auxiliary pump, driven by 28 VDC from the hot battery bus, draws fluid from the bottom of the reservoir. The pump can be activated when required to provide pressure for the roll spoilers, wheel brakes, and spoiler hold down.

The AUX HYD PWR switch is a three position switch. The ON position powers the pump continuously from the hot battery bus. The OFF position deactivates the pump. The NORM position activates the auxiliary pump automatically if spoiler holddown is selected or when the main system hydraulic pressure drops below 1,200 psi.

NOTE

If low system pressure is detected, the auxiliary pump activates even without normal DC power on the aircraft. When the engines (and APU) are shut down, the AUX HYD PWR switch must be placed to OFF to prevent battery depletion by the pump.

APU-DRIVEN PUMP

An APU with hydraulics (if installed) drives a variable displacement pump that pressurizes the entire hydraulic system. The pump is

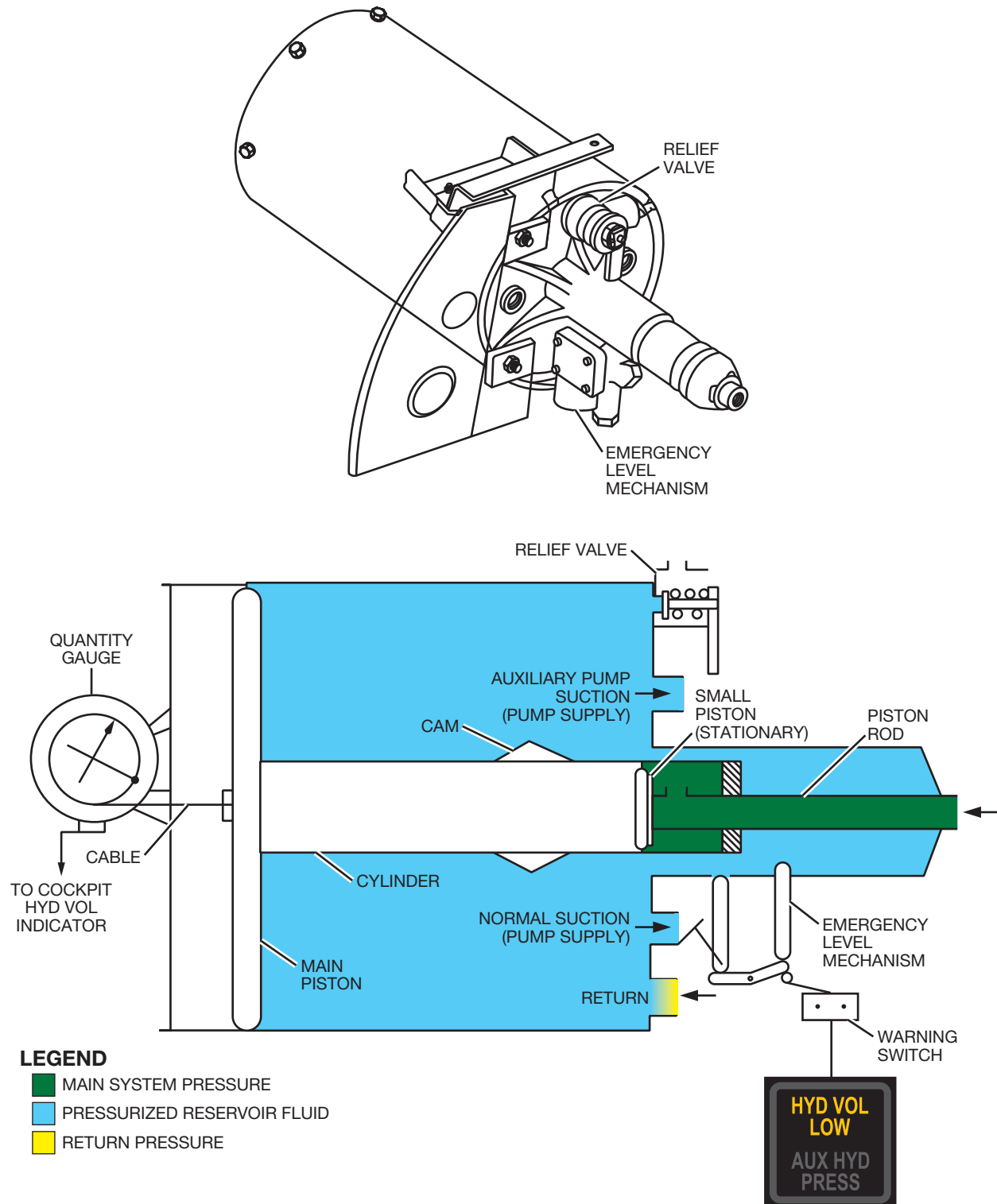


Figure 13-5. Reservoir System Schematic



identical to the pumps on the engines. If the APU pump is the only pump supplying the system, its pressure output is displayed on the cockpit gauge.

FIREWALL SHUTOFF VALVES

A firewall shutoff valve is in the suction line to each engine-driven hydraulic pump. The valves are electrically operated and are controlled by the ENG FIRE PUSH switchlights on the glareshield (Figure 13-1). The valves are normally open and are closed only if an engine fire occurs or during a preflight check when required.

ACCUMULATOR

An accumulator above the hydraulic reservoir (Figure 13-6) dampens pressure surges in the hydraulic system. The 1,500 psi nitrogen precharge is checked on a remote pressure gauge in the baggage compartment during preflight inspection.

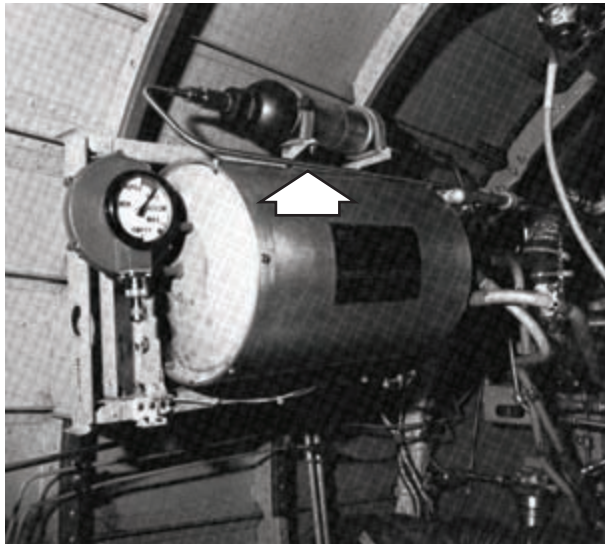


Figure 13-6. Accumulator

HYDRAULIC SUBSYSTEMS

GENERAL

Hydraulically powered subsystems include wheel brakes, spoilers, aileron boost, ECU fan (SNs 0001—0104), landing gear, thrust reversers, and nosewheel steering. Application of hydraulic power to the thrust reversers is presented in this chapter. Application of hydraulic power to the remaining subsystems is presented in Chapter 11, “Air Conditioning;” Chapter 14, “Landing Gear and Brakes;” and Chapter 15, “Flight Controls.”

THRUST REVERSERS

Description

The aircraft has electrically controlled, hydraulically actuated, target-type thrust reversers (Figure 13-7), which assist deceleration during a landing roll. When deployed, the reversers are maintained in position by hydraulic pressure.

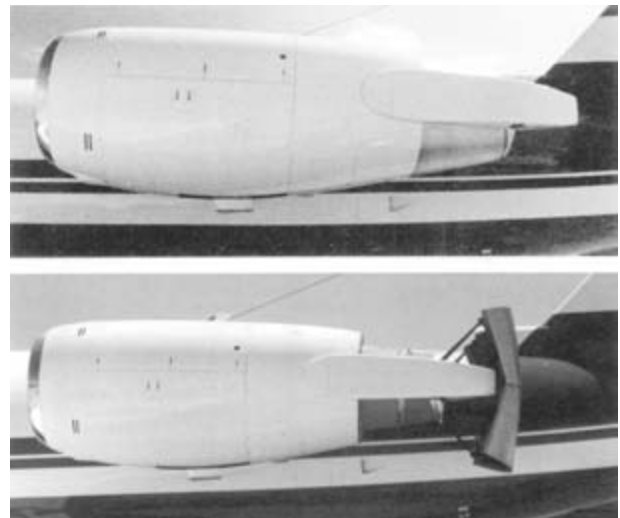


Figure 13-7. Thrust Reversers

In normal operation, hydraulic pressure is isolated when the reversers are stowed. The thrust reversers are maintained in the stowed position by an overcenter condition of the operating bar mechanism.



Thrust reverser operation is limited to ground operation only. The control circuitry is wired through the squat switches on the left and right main landing gear. Reverser levers, piggybacked on the throttles, are used to control the thrust reversers (Figure 13-8). Placing the levers down to the stow detent stows the reversers; placing the levers up to the idle detent deploys the reversers. Moving the levers above the idle reverse position increases engine thrust proportional to reverser lever movement.

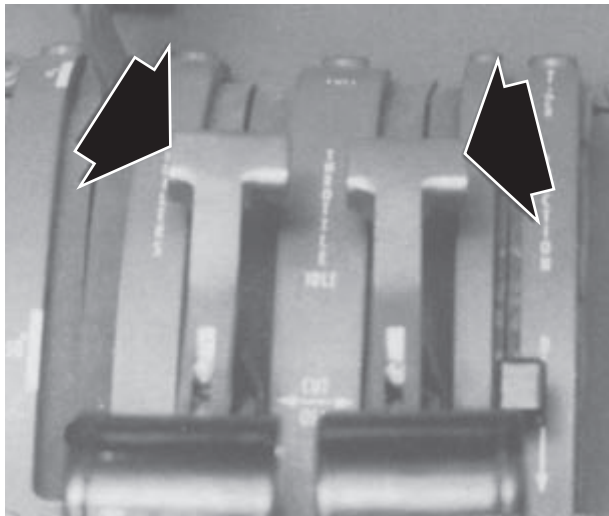


Figure 13-8. Thrust Reverser Levers

When a reverser lever is moved to the reverse idle (deploy) position, a solenoid lock prevents further aft movement until the reverser fully deploys.

A microswitch in the throttle quadrant provides electrical control of the thrust reverser. When the reverser lever is moved from the stow position, the switch closes, energizing the thrust reverser isolation valve open and energizing the reverser control valve to the deploy position (Figure 13-9). This action occurs only if a ground is completed through either main gear squat switch.

Each reverser has three lights on the glareshield panel (Figure 13-9):

- ARM
- UNLOCK
- DEPLOY

The amber ARM light circuit is completed by a pressure switch, indicating that the isolation valve is open and hydraulic pressure is available to the reverser control valve. The control valve opens if either or both main gear squat switches are in the ground mode.

The amber UNLOCK light is completed by a microswitch which closes when the reverser mechanism initially moves from the stowed position.

The white DEPLOY light circuit is completed by a microswitch which closes when the reverser door mechanism reaches the fully deployed position. This also completes the circuit to energize the solenoid lock out of the way to allow the reverser lever to be brought out of the idle deploy position.

Electrical power for the left and right thrust reversers is from the left extension and right branch buses respectively.

Throttle Feedback System

The throttle feedback system ensures that engine thrust is at idle while the thrust reversers are in transit. The feedback system is made up of a positive action mechanical linkage, connected to the fuel control. During thrust reverser transition, a flexible cable drives a spring-loaded throttle control cam which follows the throttle linkage to idle, preventing the linkage from advancing the power setting until the cycle is complete. If the thrust reverser deploys inadvertently, the feedback system reduces thrust on the affected engine to idle during the transition cycle.

CAUTION

Do not attempt to move the reverser levers beyond idle deploy until the DEPLOY lights illuminate, and do not attempt to move the throttles out of IDLE until the UNLOCK lights extinguish after stowing the thrust reversers. In the event that either occurs, maintenance is required prior to the next flight.

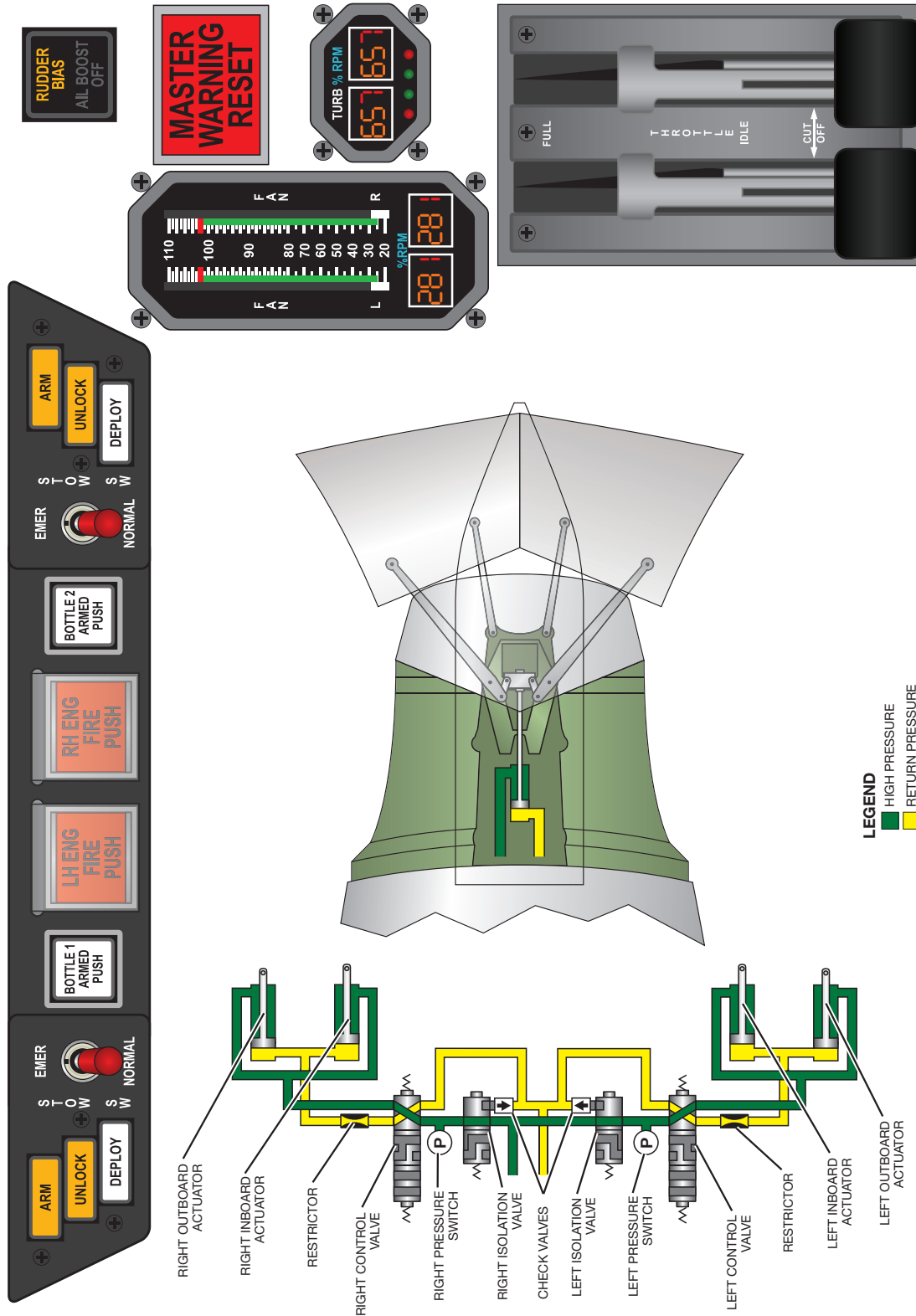


Figure 13-9. Thrust Reverser System



Operation

After landing, when the throttles are at IDLE and the nosewheel is on the ground, the thrust reverser levers can be raised to the IDLE deploy detent. The ARM lights illuminate, followed immediately by the illumination of the UNLOCK and then the DEPLOY lights. The reverser lever solenoid locks release, and the reverser levers can then be moved aft to increase thrust as desired. A mechanical interlock in the pedestal holds the throttles in IDLE. While the reverser levers are out of the stow position, the ARM, UNLOCK, and DEPLOY lights remain illuminated.

As the aircraft decelerates toward 65 KIAS, decrease engine power to achieve idle reverse at 65 knots. The thrust reversers may remain deployed below 65 KIAS to assist further deceleration by thrust attenuation and drag.

To stow the reverser, move the reverser lever fully forward and down. The control valve energizes to the stow position, which directs hydraulic pressure to the stow side of the reverser actuators. The DEPLOY lights extinguish, followed almost immediately by the UNLOCK and ARM lights, indicating that the reverser doors are in the fully stowed position.

NOTE

The RUDDER BIAS annunciator remains illuminated whenever either or both reverser levers are out of the stow position. This indicates reduced effectiveness of the rudder bias actuator, which is desirable during reverser operation.

Emergency Stow

An emergency stow system augments the normal control circuit. The system is used in case of an inadvertent deployment in flight.

A two-position STOW switch for each reverser is inboard of the reverser lights. The switch positions are EMER and NORM.

Moving a STOW switch to the EMER position energizes the isolation valve open and causes the control valve to energize to the stow position, regardless of squat switch modes. The valves, in those positions, direct hydraulic pressure to the stow side of the actuator until the switch is placed in NORM.

The ARM light remains illuminated while the emergency STOW switch is in EMER. The system is checked before flight. To check the system, place the emergency STOW switches in EMER and verify that the DEPLOY and UNLOCK lights extinguish. Place the reverser levers down, and position the emergency stow switches to NORM. The ARM lights then extinguish. If the test is not successfully completed, flight is prohibited.

If either an ARM or UNLOCK light illuminates in-flight, the MASTER WARNING RESET switchlights also illuminate.

NOTE

Electrical power for the emergency STOW switches is from the opposite side bus via the opposite thrust reverser CB. For example, the left STOW switch is powered from the right branch bus via the right thrust reverser CB.

LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

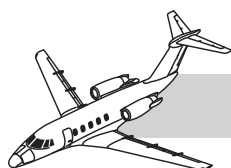
EMERGENCY/ ABNORMAL

For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.



QUESTIONS

1. Operation of the auxiliary hydraulic pump is indicated by:
 - A. Illumination of the AUX-HYD PRESS annunciator
 - B. A pressure indication on the HYD PRESS indicator only
 - C. Illumination of the HYD PRESS LOW annunciator
 - D. None of the above
2. The auxiliary hydraulic pump pressurizes:
 - A. The entire hydraulic system
 - B. All subsystems except the thrust reversers
 - C. The roll spoilers and the wheel brakes
 - D. The reservoir only
3. Reservoir fluid quantity is displayed:
 - A. Only on a gauge on the aft end of the reservoir
 - B. On the HYD VOL indicator in the cockpit
 - C. Only on the HYD VOL indicator in the cockpit
 - D. On a dipstick
4. The optional APU-driven pump provides pressure for:
 - A. Only landing gear
 - B. Only the aileron boost and speedbrake systems
 - C. All subsystems
 - D. Only the roll spoilers and the wheel brakes
5. Servicing of the hydraulic reservoir requires:
 - A. Equipment such as a hydraulic mule or a hand-operated pump
 - B. No special equipment
 - C. Electrical power applied to the aircraft
 - D. Operation of the auxiliary hydraulic pump
6. The subsystem(s) that cannot be actuated with the auxiliary pump is/are:
 - A. Landing gear
 - B. Aileron boost
 - C. Thrust reversers
 - D. All of the above
7. The thrust reversers:
 - A. May be deployed only when the throttles are in IDLE
 - B. Must have both emergency STOW switches in EMER for takeoff to guard against inadvertent deployment during that critical phase of flight
 - C. May be left in idle reverse until the aircraft is brought to a full stop
 - D. Both A and C
8. When normal deployment of the thrust reversers is obtained, the following annunciators should be illuminated:
 - A. ARM, UNLOCK, DEPLOY
 - B. DOOR UNLOCKED, ARM, UNLOCK, DEPLOY
 - C. ARM, UNLOCK, DEPLOY, RUDDER BIAS
 - D. DOOR UNLOCKED, DEPLOY

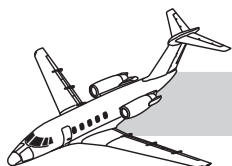


CHAPTER 14

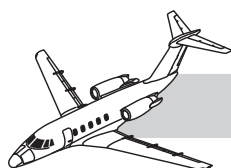
LANDING GEAR AND BRAKES

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CHAPTER 14

LANDING GEAR AND BRAKES



INTRODUCTION

The Citation 650 series aircraft landing gear is electrically controlled and hydraulically actuated. When retracted, the nose gear and struts of the main gear are enclosed by mechanically actuated doors. The main gear wheels remain uncovered in the wheel wells. Gear position and warnings are indicated by green and red lights and a warning horn. Nosewheel steering is manually controlled, hydraulically actuated, and electrically disabled. Shimmy dampening is a feature of the power steering system. Power braking with antiskid protection is provided with emergency pneumatic braking as a backup. Automatic braking occurs during gear retraction.

GENERAL

The main landing gear retracts inboard; the nose gear retracts forward. The gear is held in the up position by mechanical uplocks. When extended, the gear is mechanically locked down. Main system hydraulic pressure is applied to the extend side of the actuators as a backup.

An electrically positioned control valve directs hydraulic pressure for gear operation. Gear position is indicated by a red and three green lights on the landing gear control panel. A warning horn sounds when the throttle, flap, and gear positions are incompatible.



The hydraulically actuated nosewheel steering system is controlled by mechanical linkage from the rudder pedals and the steering tiller on the pilot console. The nose gear is centered mechanically when the gear extends at liftoff.

The power brake system includes a multi-disc brake assembly in each main wheel. Pressure is normally provided by the main hydraulic system or by the auxiliary hydraulic pump. Antiskid protection permits maximum braking without wheel skid under varying runway conditions. Emergency pneumatic braking without antiskid protection is available from an emergency nitrogen bottle.

LANDING GEAR

DESCRIPTION

The main and nose gear shock struts are conventional air/oil struts. Each strut has a floating piston with hydraulic fluid on one side and a nitrogen pressure charge on the other side for shock absorption during taxi, takeoff, and landing. The landing gear is normally hydraulically actuated but can be mechanically and pneumatically extended if the hydraulic gear actuation fails.

COMPONENTS

Main Gear Assembly

The main gear assembly includes (Figure 14-1):

- Solid strut
- Shock strut
- Trailing link with axle-mounted dual wheels
- Squat switch
- Actuator
- Uplock assembly
- Side brace
- Gear door

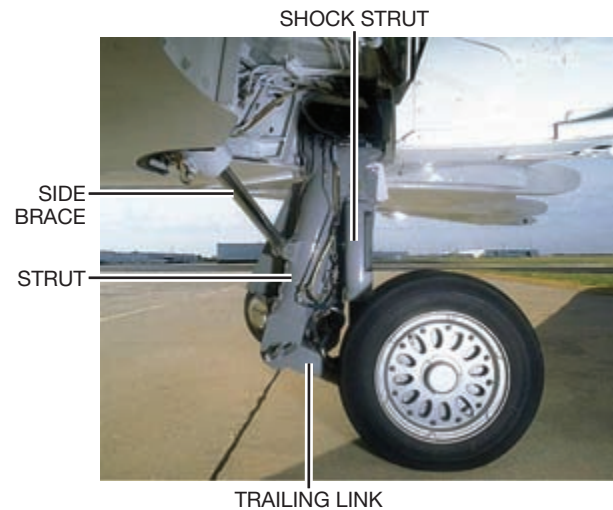


Figure 14-1. Right Main Gear and Door

Each wheel contains a multi-disc brake assembly. The trailing link is attached to, and trails behind, the base of the solid strut.

The side brace includes the mechanical downlock. The actuator exerts extension/retraction force on the top side of the solid strut.

When the gear is extended, it is locked down by spring-loaded locks within the side brace. The gear is released only with hydraulic pressure applied to the side brace. The main gear uplock mechanism is hydraulically armed for retraction and hydraulically disengaged for extension. A door, actuated by gear movement, covers the main gear strut when retracted.

Each main gear wheel incorporates three fusible plugs, which melt and deflate the tire if excessive wheel temperature is generated by an overheated brake.

Nose Gear Assembly

The nose gear assembly includes (Figure 14-2):

- Shock strut
- Retract/extend actuator with integral downlock
- Uplock

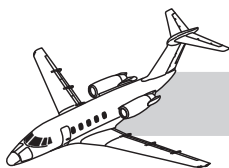


Figure 14-2. Nose Gear and Doors

- Torque links
- Single wheel
- Squat switch
- Centering switch
- Shimmy damper
- Power steering unit atop the strut
- Gear doors

When the nose gear is extended, an internal locking mechanism in the gear actuator engages, locking the gear down. The locking mechanism is released only with hydraulic pressure applied to the retract side of the actuator.

The nose gear is locked in the retract position by a spring-loaded hook. The uplock is hydraulically released for extension. Three doors are mechanically actuated by strut movement to enclose the gear and wheel on retraction. The two forward doors and aft spade door remain open with the gear extended.

At liftoff, the nose gear is centered by a centering cam, which maintains nosewheel alignment during extension and retraction.

CAUTION

Do not tow or taxi the aircraft with the strut extended more than 7 inches.

CONTROLS AND INDICATIONS

Landing Gear Control Handle

The landing gear control handle is used to control the landing gear. The gear position is indicated by one red and three green lights on the control panel (Figure 14-3). A warning horn provides aural warning if the gear is not down and locked.

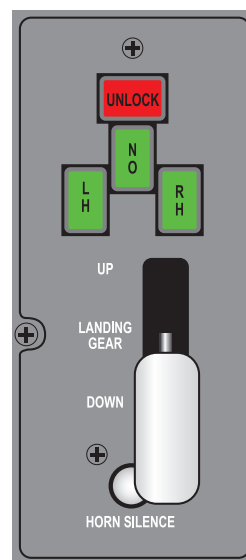


Figure 14-3. Landing Gear Control Panel

The landing gear control handle actuates switches, which complete circuits to the extend or retract solenoid of the gear control valve. A solenoid-operated, spring-loaded pawl holds the handle in the DOWN position, which prevents inadvertent movement of the handle to the UP position (Figure 14-4) when the aircraft is on the ground.

The pawl is electrically released at liftoff by actuation of one or both main gear squat switches into the flight mode, and by two nose strut switches positioned by extension and centering of the nose gear. The landing gear handle can then be moved to the UP position.

This safety feature cannot be overridden. If either electrical power or the solenoid fails, the gear handle cannot be moved to the UP

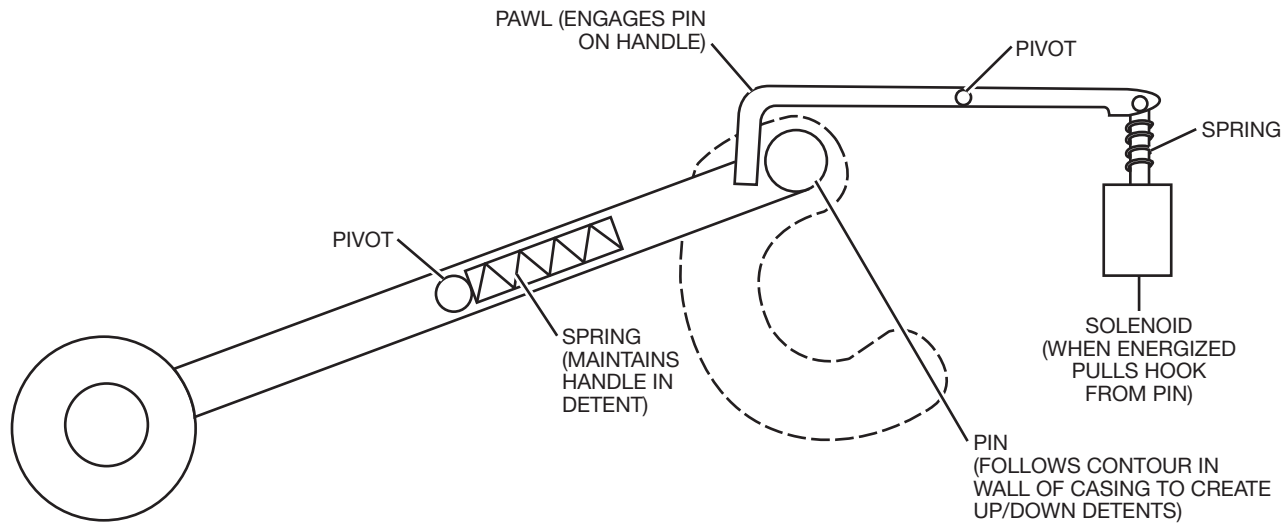


Figure 14-4. Landing Gear Handle Locking Solenoid

position. The gear handle must be pulled out of a detente position in order to move it to the UP or DOWN position.

Gear Position Indicators

The green NO, LH, and RH lights on the landing gear control panel indicate gear down and locked. As each gear locks down, its respective light illuminates.

The red UNLOCK light illuminates when the landing gear handle is moved out of the UP detent, indicating an unsafe gear position. The UNLOCK light remains illuminated until all three gears are down and locked. During retraction, the UNLOCK light illuminates when any downlock is released and remains illuminated until all three gears are up and locked.

Normal indication with the gear down is illumination of three green lights. When the gear is retracted, the lights extinguish.

Figure 14-5 shows the light indications for various gear positions. To test the UNLOCK light, the three green lights, and the warning horn, position the rotary TEST knob to LDG GR.

Warning Horn

The warning horn sounds and cannot be silenced if all gears are not down and locked and if either of the following conditions exist:

- The flaps are extended beyond 20°.
- Both throttles are retarded below 80% N₁ rpm, radar altitude is less than 500 feet AGL, and the flaps are extended beyond 7°.

The horn also sounds, but can be silenced, if all gear are not down and locked and if the following condition exists:

- One or both throttles are retarded below 55% N₁ rpm, and airspeed is below 163–170 KIAS.

To silence the horn under this condition, press the HORN SILENCE button on the landing gear control panel (Figure 14-3). To rearm the system for subsequent operation, advance either throttle.

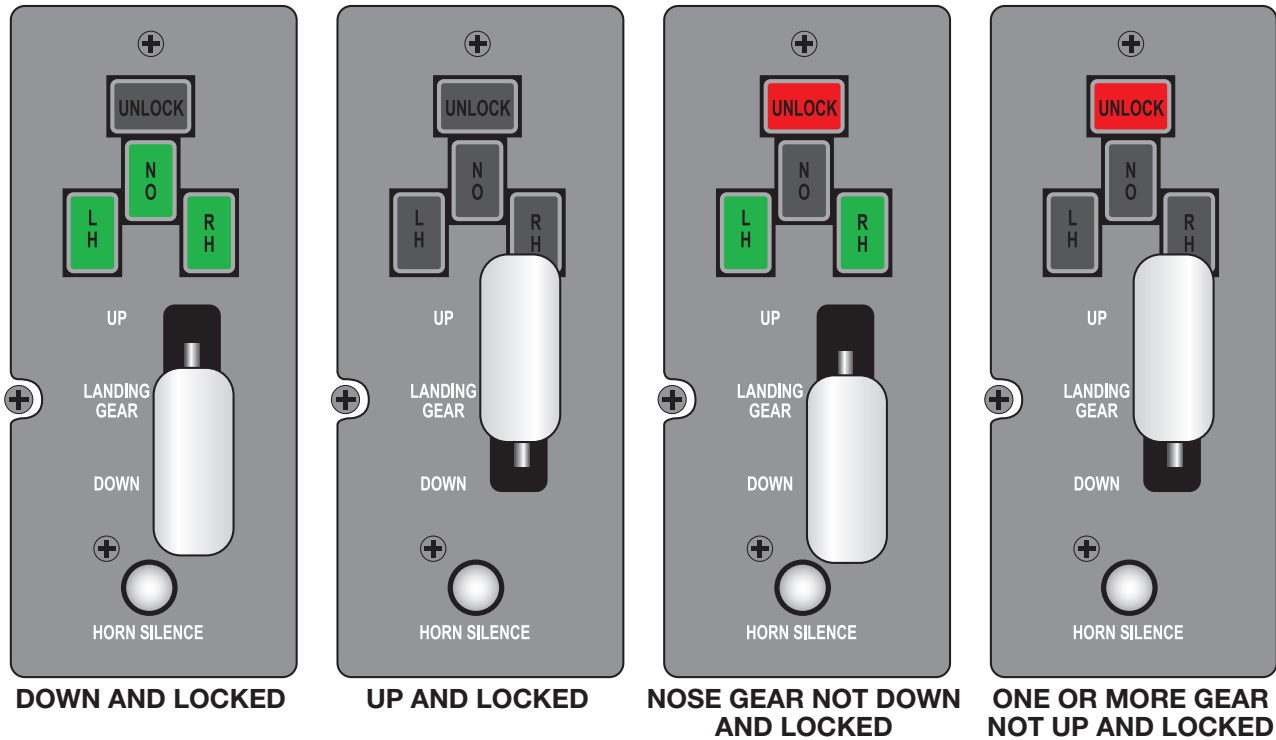


Figure 14-5. Gear Position Indications

OPERATION

Landing Gear Retraction

To retract the landing gear, place the landing gear handle to the UP position (Figure 14-6). Doing so electrically positions the gear control valve and supplies pressure simultaneously to the gear actuators, which retract the gear to the main gear uplocks, and to each side brace to release the downlocks.

As the main gear uplock roller contacts the uplock latch, the latch releases and pressure drives the uplock hook to the overcenter locked position, securing the main gear in the retracted position (Figure 14-6). The uplock hook locks the nose gear in the UP position by mechanical action.

As the gear retracts, pressure is routed to the brake metering valve to stop main wheel rotation. When the gear is up and locked, retract pressure is relieved as the control valve deenergizes. The uplocks maintain the gear in the retracted position.

Landing Gear Extension

To extend the landing gear, move the landing gear handle to the DOWN position (Figure 14-7). Doing so electrically positions the control valve to supply hydraulic pressure simultaneously to the gear actuators for gear extension and uplock release.

The main gears are locked down by spring-loaded locks in the side braces, which extend mechanically as the gear extends. The nose gear is mechanically locked down by an internal lock in the nose gear actuator. Pressure, while available, is continuously applied to the down side of the gear actuators as a backup to the mechanical downlocks.

Emergency Extension

If the landing gear cannot be extended by the hydraulic system, the auxiliary system can be used. To use the auxiliary system, place the landing gear handle to the DOWN position and ensure that the gear control circuit breaker is in.



CITATION 650 SERIES PILOT TRAINING MANUAL

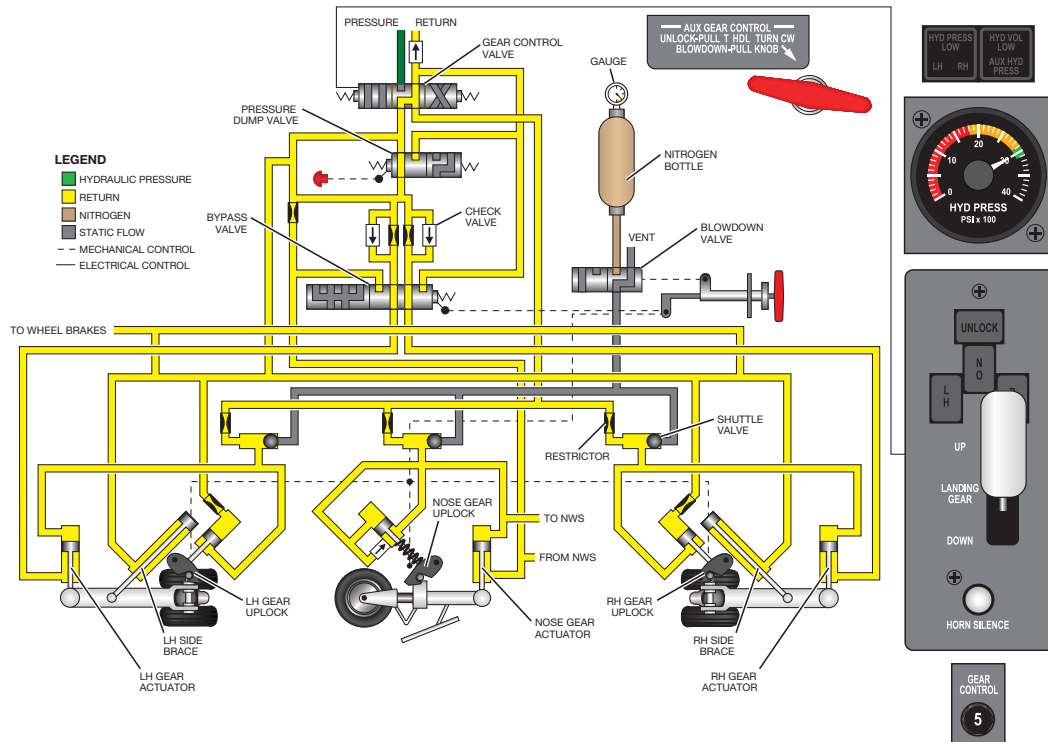


Figure 14-6. Landing Gear Retracted

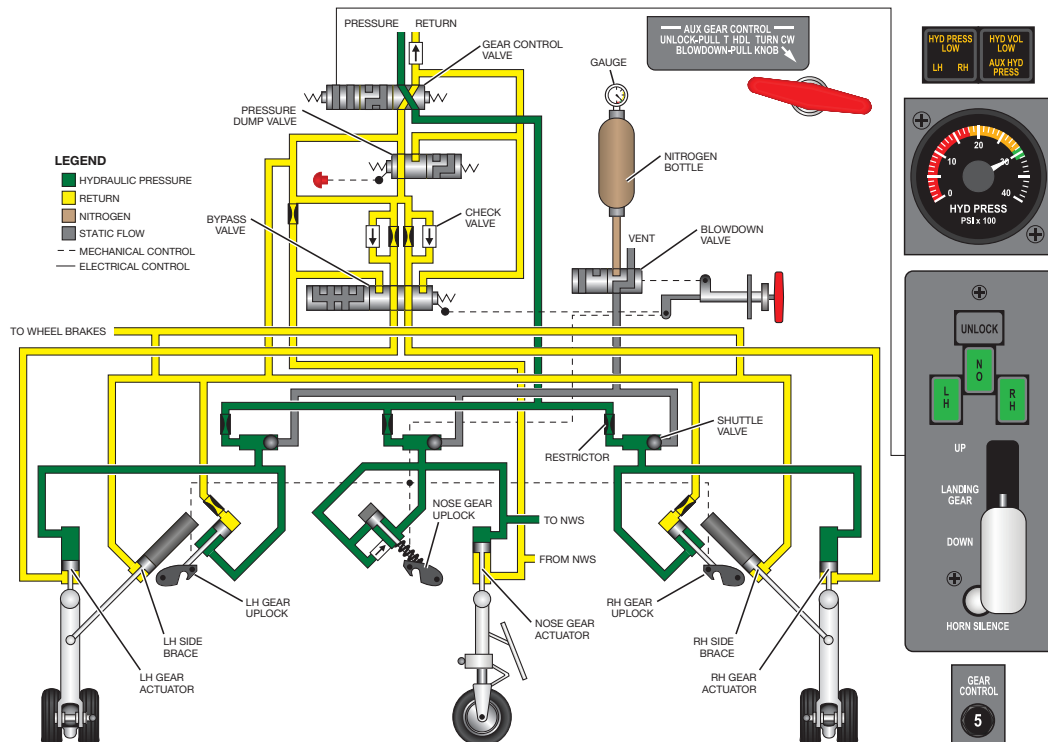


Figure 14-7. Landing Gear Extended



The AUX GEAR CONTROL T-handle (Figure 14-8) under the pilot instrument panel is pulled and locked into position by rotating it clockwise. This action mechanically releases the uplocks and actuates the mechanical bypass valve in the gear control module. The gear should free-fall to the down and locked position. Pulling the round control knob behind the T-handle opens a valve on the pneumatic bottle releasing dry nitrogen to the extend side of the gear actuators and to the uplocks. When all gears indicate down and locked, push the round control knob to the stowed position and rotate the AUX GEAR CONTROL T-handle counterclockwise and push to the stowed position.



Figure 14-8. AUX GEAR CONTROL T-Handle

If the AUX GEAR CONTROL T-handle cannot be pulled out, a hydraulic (liquid) lock may exist in the system. Actuating the pressure dump valve (potty valve) control under the vanity positions the pressure dump valve to release blocked fluid to return (Figure 14-9).

CAUTION

The landing gear must not be retracted after a free-fall until a maintenance inspection is performed.

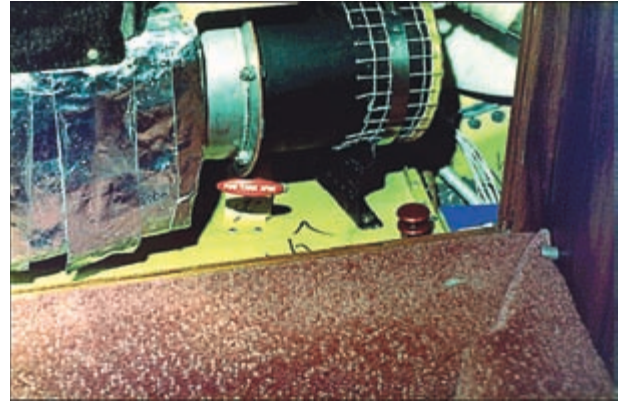


Figure 14-9. Pressure Dump Valve Control

NOSEWHEEL STEERING

DESCRIPTION

Nosewheel steering is hydraulically actuated and manually controlled through the rudder pedals and a tiller on the pilot side console. Rudder pedal steering provides 6° of nosewheel deflection either side of center. Rudder pedal input linkage is through mechanical bungees to the steering gearbox.

The tiller permits 75° of steering for taxiing and maneuvering in congested areas. The tiller has more authority than rudder pedal inputs. Additional nosewheel steering beyond either 6° or 75° is available with differential braking. Maximum deflection is 90°.

The nosewheel steering unit is on the nose gear strut and is a rack and pinion hydraulic actuator. The nosewheel steering unit dampens shimmy with assistance from the friction brake.

CONTROLS AND INDICATIONS

Nosewheel steering controls are shown in Figure 14-10.

OPERATION

Pressure from the main hydraulic system is applied to the directional control valve in the power steering unit and to the accumulator. (Figure 14-11).

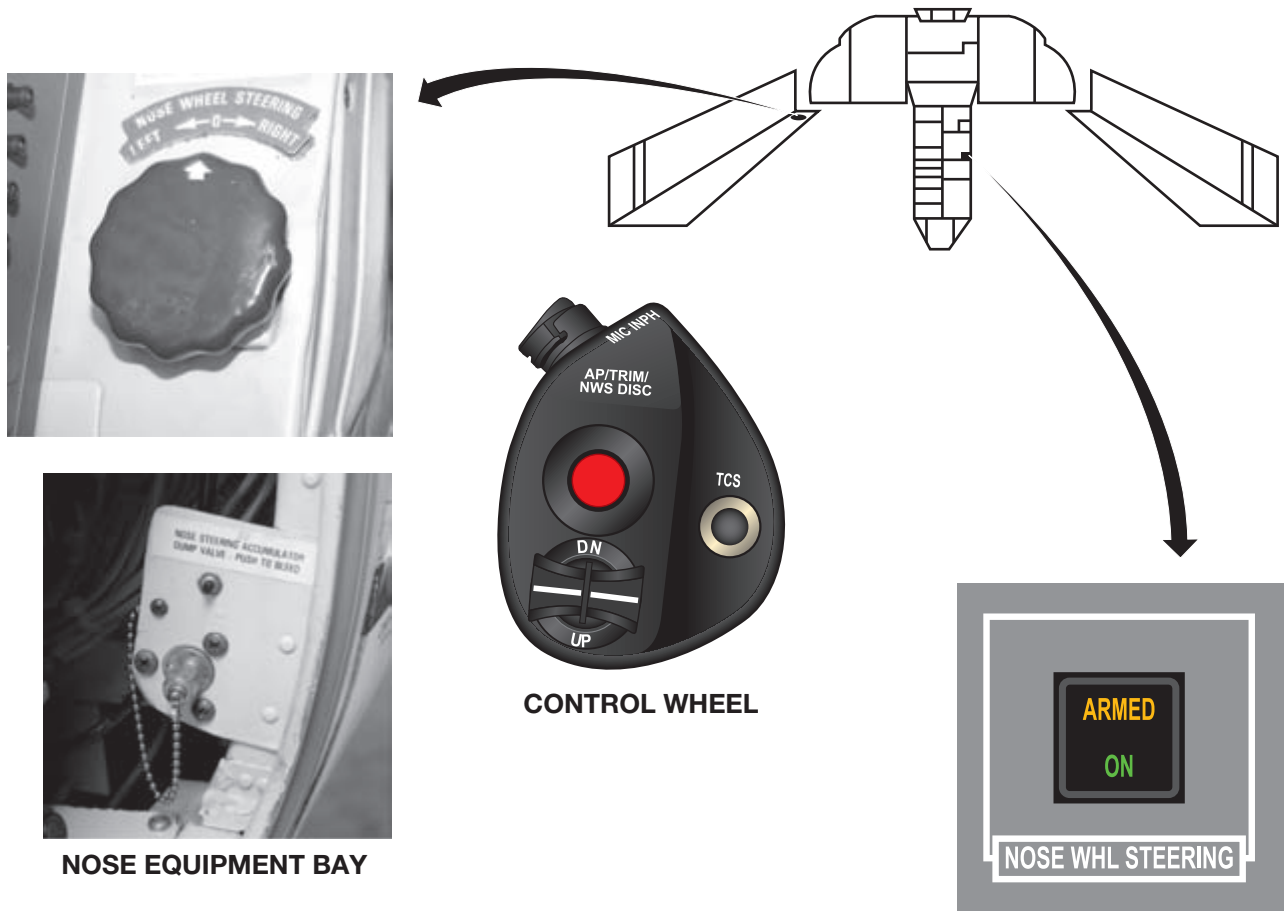


Figure 14-10. Nosewheel Steering Controls

Steering inputs are provided either through the rudder pedals or from the NOSE WHEEL STEERING tiller on the pilot side console. Steering inputs are transmitted mechanically to the directional control valve. As the valve is displaced, pressure is routed to the bypass valve, which deenergizes open when the NOSE WHL STEERING switchlight is ON (Figure 14-10). The bypass valve is spring-loaded open if normal DC power is lost.

As the steering actuator deflects the nosewheel to the appropriate position, mechanical feedback repositions the directional control valve to neutral, and steering deflection ceases until the control linkage is moved again.

The nosewheel steering system is controlled by a switchlight on the pedestal, which has an

amber ARMED position and a green ON position (Figure 14-10). When the aircraft powers up, the control valve energizes closed and the switchlight extinguishes.

Pressing the switchlight deenergizes the bypass valve and illuminates the green ON light. During ground operation the valve can be energized closed by pressing the switchlight or the red AP/TRIM/NWS DISC button on the outboard horn of either control wheel.

At rotation, when the nose gear squat switch moves to the airborne position, the valve is energized closed, the green ON light extinguishes, and the amber ARMED light illuminates. When the nose gear downlock releases during retraction, the ARMED light extinguishes.

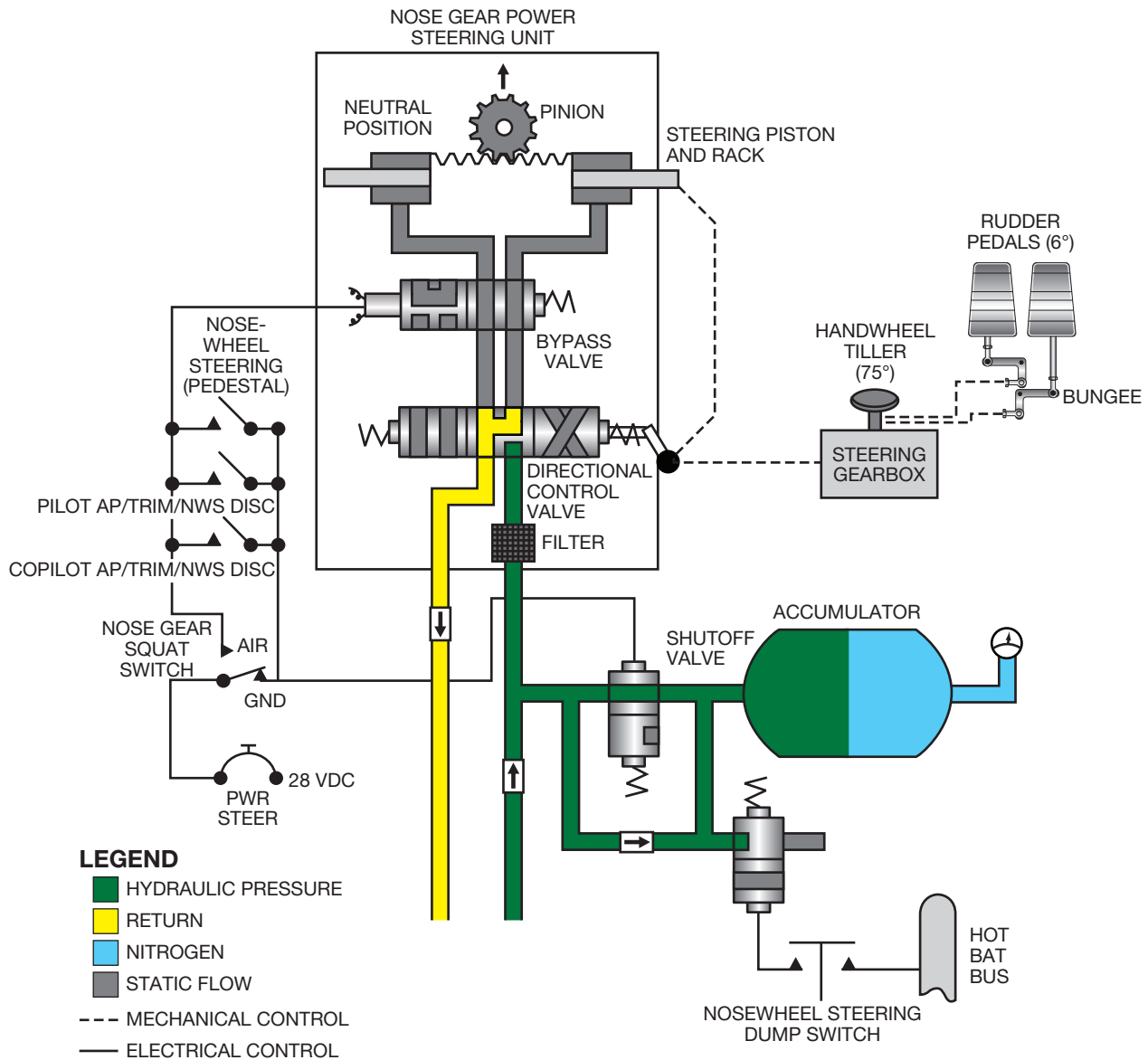
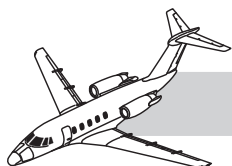


Figure 14-11. Nosewheel Steering Schematic

After the nose gear indicates down and locked prior to landing, pressing the NOSE WHL STEERING switchlight illuminates the ARMED light. In-flight, pressing the red AP/TRIM/NWS DISC button does not disarm nosewheel steering.

With the system armed and the nose gear squat switch indicating that the aircraft is on the ground the ARMED light extinguishes, and the ON light illuminates and the valve deenergizes open.

Upon touchdown, the nose gear squat switch opens a solenoid valve, allowing the nose-wheel steering accumulator to provide pressure to the steering system if the hydraulic system fails. The accumulator, which is in the nose equipment area, has a solenoid bleed valve that allows checking of the nitrogen precharge during preflight inspection. The valve is powered from the hot battery bus and is operated by a pushbutton switch (Figure 14-10) next to the oxygen filler port in the nose equipment bay.



CAUTION

Towing or taxiing with the nose strut extended 7 inches or more can damage the centering mechanism. Deflection of the nosewheel beyond 90° during towing damages the steering actuator.

When either the main or auxiliary hydraulic system supplies pressure, the hydraulically actuated power brakes are available (Figure 14-12).

CONTROLS AND INDICATIONS

The desired mode is selected with the ANTI SKID switch which has three positions ON–OFF–TEST (Figure 14-12).

BRAKES

DESCRIPTION

The aircraft is equipped with a Hydro Aire Mark II antiskid brake system with individual wheel control. The system can operate in either power brake or antiskid-protected power brake mode.

OPERATION

The brakes are operated by the power brake valve, which is connected by cables to the brake pedals. Separate cables are connected to the pilot and copilot pedals for individual brake control. If the pilot and copilot brakes are applied simultaneously, the pilot exerting the greater force has control of the brakes.

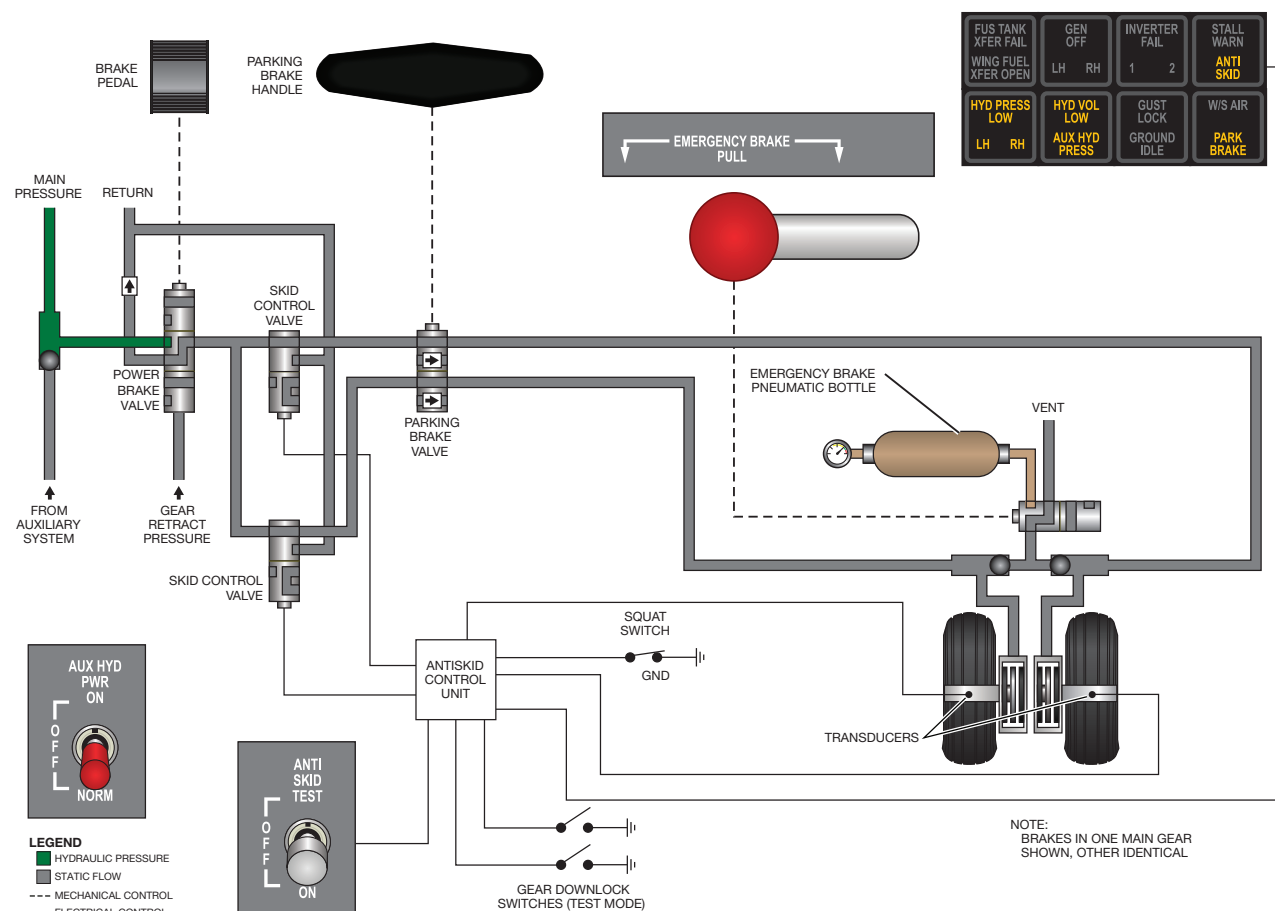


Figure 14-12. Brake System Schematic



ANTISKID CONTROL SYSTEM

DESCRIPTION

The antiskid control system prevents wheel skid under all weather conditions. Antiskid features include automatic braking upon gear retraction, touchdown and hydroplaning protection, locked wheel protection, and built-in test capability, including continuous monitoring and pilot-initiated dynamic tests.

The wheel-driven transducers generate signals based on wheel speed. If sudden deceleration occurs, which indicates a potential skid, then a signal is transmitted to the control box. The control box commands the antiskid valve to reduce pressure to the appropriate brake, permitting the wheel to return to speed. Upon acceleration, another signal is transmitted to the antiskid valve, commanding an increase in brake pressure. The complete cycle occurs many times per second to maintain braking efficiency.

COMPONENTS

The antiskid system consists of the following components:

- Four axle-mounted transducers driven by the wheels
- Electrical control box
- Power brake valve
- Two dual antiskid servo valves
- Associated switches, annunciators, and circuit breakers

The skid control transducers are linear, permanent magnet DC generators, bolted in the main gear axles and driven by each wheel. Each transducer generates a signal that is transmitted to the control box.

The control box contains electronic components on a printed circuit board. The components form four separate control circuits, one

for each transducer. Individual signals from the transducer are compared within the electronic circuitry and an appropriate signal is transmitted to the antiskid servo valve(s).

The antiskid control valves modulate the pressure furnished by the power brake valve. If a skid occurs, the control box signals the appropriate valve to reduce brake pressure, allowing the wheel to return to normal speed.

CONTROLS AND INDICATIONS

The ANTISKID switch activates the antiskid system (Figure 14-12). When the aircraft is on the ground and the antiskid circuit breaker is engaged, the built-in test activates the ANTISKID annunciator under the following conditions:

- ANTISKID switch OFF, loss of power to the control box, or failure of the power regulator circuits in the control box.
- Circuit open or shorted in one or more wheel speed transducers or antiskid control valves.
- System dynamic test initiation (6 second illumination is normal). The ANTISKID annunciator remains illuminated if the dynamic test fails.
- In-flight, when the gear is extended and the ANTISKID switch is ON, the annunciator illuminates for 3 seconds during the test, then extinguishes if no faults are found.

OPERATION

Positioning the ANTISKID switch ON activates the antiskid system, which provides automatic skid control if speed is greater than 12 knots.

TOUCHDOWN PROTECTION

The right and left squat switches provide touchdown protection for the inboard and outboard brakes respectively. Pressure to the brakes is blocked prior to touchdown and wheel spin-up by a full brake pressure release signal to all



antiskid valves. As an additional safety feature, a sustained brake pressure release is available for 5 seconds after touchdown if the wheels have not spun up above 35 knots.

The spin-up override feature allows normal antiskid braking if both squat switches fail in the airborne mode. The override becomes ineffective as the aircraft decelerates through 35 knots, at which point the affected brakes are released by the airborne squat switch signal to the control box.

If brake loss occurs, position the ANTISKID switch to OFF to restore normal power braking. The airborne signal from the squat switches is inhibited when both main gears are unlocked during retraction, allowing anti-spin braking through the brake metering valve.

LOCKED WHEEL PROTECTION

Locked wheel protection is available above 40 knots and as an alternate method of wheel protection if a primary control circuit fails.

The inboard wheels are locked wheel mates, as are the outboards. If a wheel slows to less than 50% of its mate, the locked wheel feature is activated by the fast wheel, and a full brake release signal is sent to the slow wheel's antiskid valve. The signal persists as long as the 50% speed difference exists, or until the aircraft slows below 40 knots. Normal braking operation of the fast wheel is not affected by the released wheel.

EMERGENCY BRAKES

OPERATION

If the hydraulic system fails or if the HYD VOL LOW annunciator illuminates, to restore normal braking with antiskid, activate the auxiliary hydraulic pump using the AUX HYD PWR switch.

If the auxiliary hydraulic system fails or is not selected, a pneumatic system is available (Figure 14-13).

The nitrogen used for pneumatic operation of the brakes is in an air bottle on the left forward side of the forward pressure bulkhead.

To activate emergency braking pull the red EMER BRAKE handle located left of the AUX GEAR CONTROL T-handle (Figure 14-8). Doing so applies equal pneumatic pressure to all four wheel brakes.

Releasing the EMER BRAKE handle forward releases the brakes by venting residual air overboard. Since differential braking is not possible, nosewheel steering must be used for directional control. Antiskid is not operable during emergency braking.

For the most efficient use of the emergency brakes, apply sufficient pressure to maintain the desired deceleration rate until the aircraft stops. Repeated applications rapidly deplete the pneumatic bottle.

Do not attempt to taxi with the emergency brakes. Do not attempt to apply normal and emergency brakes simultaneously; otherwise nitrogen may enter the hydraulic system.

PARKING BRAKES

DESCRIPTION

The parking brakes, which are part of the normal brake system, use valves to prevent fluid return after the brake pedals are released.

OPERATION

The parking brakes require hydraulic system pressure from either the engine driven hydraulic pumps or the auxiliary hydraulic pump. To set the parking brakes, press and hold the brake pedals and then pull the parking brake handle located aft of the center console (Figure 14-14).

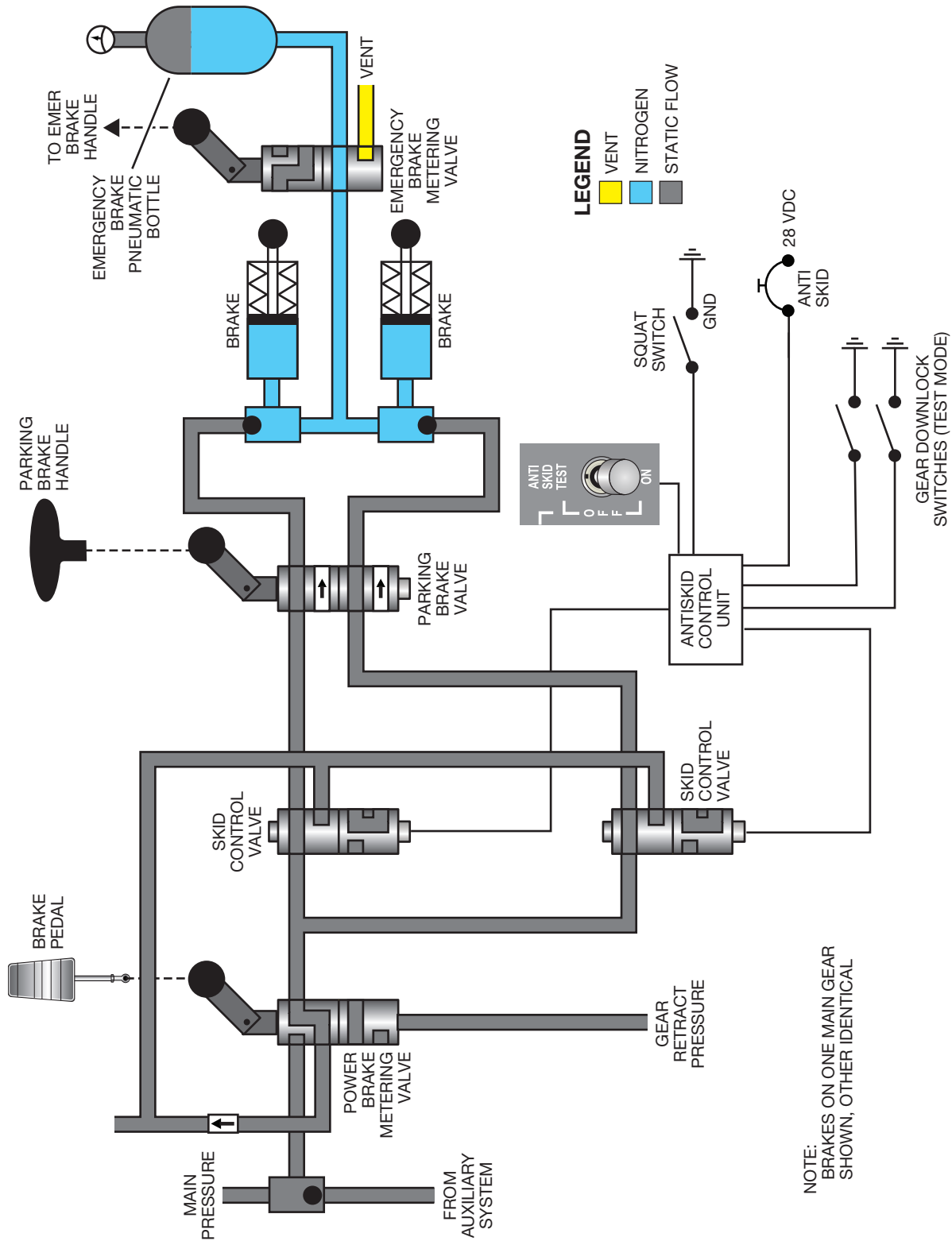


Figure 14-13. Emergency (Pneumatic) Braking



Figure 14-14. Parking Brake Handle

Do not set the parking brakes if the brakes are hot. Doing so increases brake cooling time because of decreased air circulation and may result in heat transfer from the brakes, which may open the thermal relief valves or melt the thermal fuse plugs in the wheels.

When set, the parking brakes disable the antiskid function. The parking brakes must be off during all normal braking.

When the valves are not completely open, the PARK BRAKE annunciator illuminates

LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

EMERGENCY/ ABNORMAL

For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.

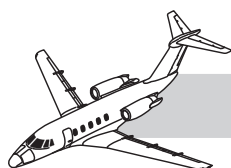


QUESTIONS

1. On the ground, the landing gear handle is prevented from movement to the UP position by:
 - A. Mechanical detents
 - B. A spring-loaded locking solenoid
 - C. Hydraulic pressure
 - D. A manually applied handle locking device
2. Concerning the landing gear, the correct statement is:
 - A. Hydraulic pressure is removed from the actuators when the uplocks or downlocks are engaged
 - B. Hydraulic pressure is applied to the main landing gear downlocks during gear extension
 - C. Hydraulic pressure is applied to the nose gear uplock during extension and retraction
 - D. Hydraulic pressure is applied to the main gear uplocks during extension and retraction
3. Landing gear downlocks are disengaged:
 - A. When hydraulic pressure is applied to the side braces on the main gear
 - B. By action of the gear squat switches
 - C. By removing the external downlock pins
 - D. By mechanical linkage as the gear begins to retract
4. Each main gear wheel incorporates fusible plugs that:
 - A. Blow out if the tire is over serviced with air
 - B. Melt, deflating the tire if an overheated brake creates excessive wheel temperature
 - C. Are thrown out by centrifugal force if maximum wheel speed is exceeded
 - D. None of the above
5. Upon retraction, if only the nose gear does not lock in the UP position, the gear panel annunciation is:
 - A. Red annunciator on, green LH and RH annunciators on.
 - B. Red annunciator extinguished, green LH and RH annunciators on
 - C. Red annunciator illuminated, and all three green annunciators extinguished
 - D. All four annunciators off
6. Sounding of the gear warning horn cannot be silenced when one or more gears are not down and locked and:
 - A. Flaps are extended beyond the 20° position
 - B. Both throttles are retarded below 80% N₁ rpm, radar altitude is less than 500 feet AG, and the flaps are extended beyond 7°
 - C. Either throttle is retarded below 55% N₁ rpm
 - D. Both A and B
7. Concerning the landing gear emergency extension system, the correct statement is:
 - A. The landing gear handle is placed to DOWN to release the uplocks
 - B. This system must be used if a complete electrical failure occurs
 - C. All controls for this system are in the cockpit
 - D. The uplocks can only be released by pulling the AUX GEAR CONTROL T-handle
8. Nosewheel steering is operative:
 - A. Only on the ground with normal DC power available
 - B. With the gear extended or retracted
 - C. With the gear extended, in-flight or on the ground under all conditions
 - D. None of the above



9. The power brake valve is actuated:
- A. Mechanically by the brake pedals
 - B. Mechanically by the emergency air-brake control lever
 - C. Hydraulically by master cylinder pressure
 - D. Automatically at touchdown
10. When the emergency brakes are used:
- A. The handle should be pumped to build up sufficient pressure to stop the aircraft
 - B. The toe brakes must be applied to allow pneumatic pressure to reach the brakes
 - C. Differential braking is not available
 - D. Reduced braking is available if the landing gear has been extended pneumatically
11. Concerning the landing gear, the correct statement is:
- A. The red UNLOCK annunciator illuminates and the warning horn sounds when either or both throttles are retarded below 55% N_1 rpm and the gear is up
 - B. The gear warning horn can be silenced when the gear is not down and locked and the flaps are extended beyond 20°
 - C. The landing gear pins must be inserted on the ground due to loss of hydraulic pressure as the engines are shut down
 - D. The landing gear is secured in the extended position by mechanical locks
12. The parking brakes:
- A. May be set immediately after a maximum braking effort due to the modulation of the antiskid system
 - B. Will still be operable if the emergency brakes have to be utilized
 - C. Cannot be set if auxiliary hydraulic pressure is being used for braking
 - D. None of the above
13. The power brake system:
- A. Is powered only by the main hydraulic system
 - B. Is powered only by the auxiliary hydraulic system
 - C. Can be powered by either the main or auxiliary hydraulic system
 - D. Does not have an emergency backup system except the auxiliary hydraulic pump
14. With regard to the antiskid system:
- A. The system operates even though the brakes are applied prior to touchdown
 - B. The ANTI SKID annunciator illuminates only when the ANTI SKID switch is OFF
 - C. Manual braking can be achieved by placing the ANTI SKID switch in OFF and pumping the brake pedals
 - D. The ANTI SKID switch must be placed in OFF for taxiing

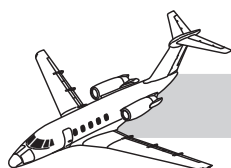


CHAPTER 15

FLIGHT CONTROLS

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CHAPTER 15

FLIGHT CONTROLS



INTRODUCTION

The primary flight controls of the Citation 650 series aircraft consist of ailerons, rudders, and elevators. The ailerons are hydraulically boosted; the rudder and elevators are manually actuated by rudder pedals and conventional control columns. Rudder and aileron trim is mechanical, and pitch trim is electrical.

Secondary flight controls consist of trim systems, electric flaps, and hydraulic spoilers. Eight spoiler panels on the upper wing can be used as speedbrakes to assist in lateral control, for emergency descent, and for supplemental braking after touchdown.

GENERAL

The rudder and elevators are manually operated by conventional rudder pedals and control wheels. Control inputs are transmitted to the control surfaces through cables and bellcranks. The rudder pedals can be adjusted to three separate positions with a spring-loaded latch on the rudder pedal.

Lateral control is provided by hydraulically boosted ailerons, assisted by roll spoilers.

Eleven vortex generators are positioned on the top surface of the wing forward of the outboard roll spoiler panel.



Figure 15-1 shows all flight control surfaces, including primary, secondary, and trim tabs.

PRIMARY FLIGHT CONTROLS

AILERON SYSTEM

Lateral control is provided by the combined action of the ailerons and the roll spoilers. Rotation of the control wheels provides control inputs to both systems. Aileron effectiveness is enhanced by roll spoiler operation occurring simultaneously with up-aileron deflection on the down wing. A spring cartridge in the control linkage provides an artificial feel force. If hydraulic pressure is lost, the ailerons revert to manual operation.

Figure 15-2 shows the aileron control cable routing.

Operation

The mechanical linkage to the ailerons is conventional. The aileron hydraulic actuator is on the quadrant in the rear of the fuselage (Figure 15-3).

Pressure to the actuator is controlled, or enabled by a solenoid shutoff valve that is energized open by normal DC power. Control of the shutoff valve is via the AIL BOOST switch which has ON–OFF–RESET positions (Figures 15-3 and 15-4).

The aileron quadrant moves the valve on the aileron boost servo actuator through a force link assembly (Figure 15-3). The force link incorporates an internal spring between two slides, a cam, cam follower, and an override microswitch. The force link is designed to sense if excessive control force is required due to possible actuator malfunctions. If this occurs, the force link spring compresses allowing the cam to move away from the cam follower and the microswitch to deenergize a shutoff valve closed to shut off hydraulic pressure to the actuator. This prevents a hydraulic lock.

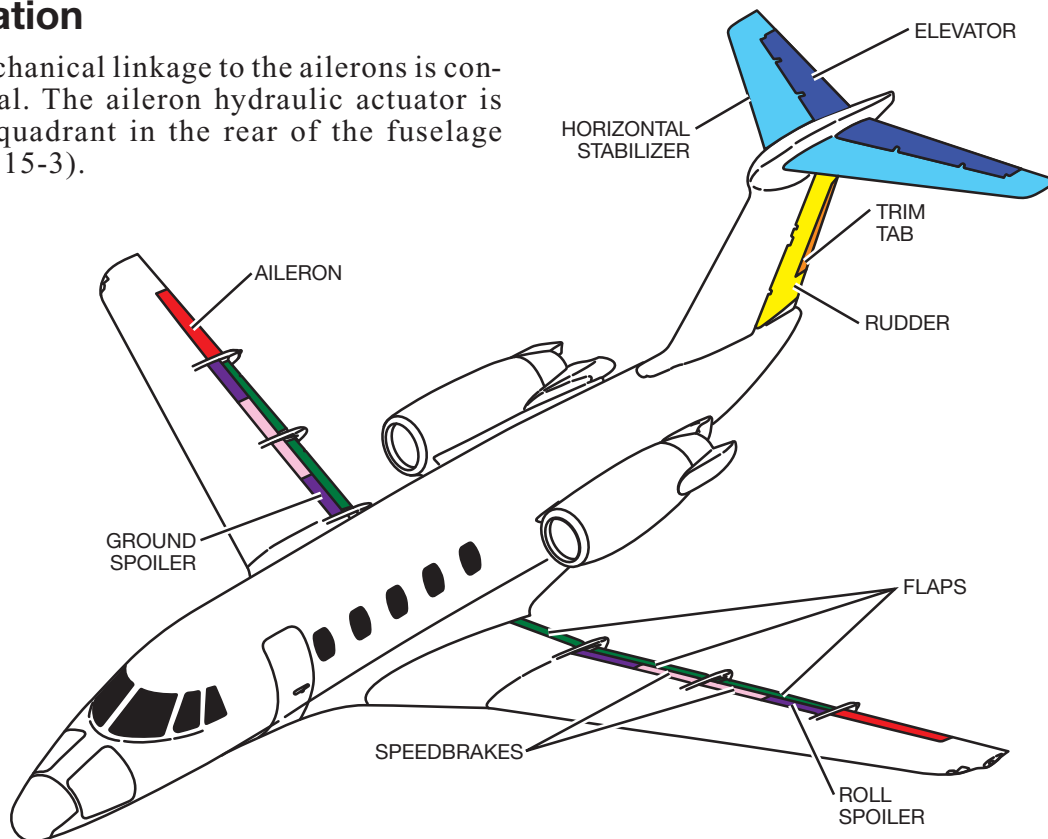


Figure 15-1. Flight Control Surfaces



To meet certification criteria, the force link must be tested. In order to allow DC power to open the valve, the force link must first be tripped (Figure 15-5) upon initial application of DC power. To trip the force link on the ground, move the control wheel to the stop position, and then release it. To trip the force link in flight, rapidly move the control wheel.

The microswitch disconnect position applies power to the powered aileron latching relay. Once energized, the relay remains closed as long as electrical power is applied to the AILERON BOOST CB on the left cockpit CB panel.

DC power through the closed powered aileron latch is available to the AIL BOOST switch. When the switch is positioned to RESET, power is applied to ground the powered aileron reset relay. Once it is initially powered, the relay remains closed until the force link is tripped or power is removed from the entire system. With the relay closed, a circuit path is made available to the shutoff valve.

When the AIL BOOST switch is positioned to ON, the valve is energized open allowing

hydraulic pressure to the servo valve on the aileron actuator.

The servo valve, mechanically operated by aileron control linkage, meters hydraulic pressure to the actuator proportional to control wheel movement. If the valve does not precisely follow the linkage, the force link micro switch trips, removing power from the aileron reset relay and shutoff valve. To restore pressure, position the AIL BOOST switch to RESET, and then to ON.

If pressure is lost to the aileron boost actuator, an amber AIL BOOST OFF annunciator illuminates and a warning horn sounds. Aileron actuation reverts to manual operation via the control linkage.

An out of sync switch between the upper and lower sections of the outer shaft also removes

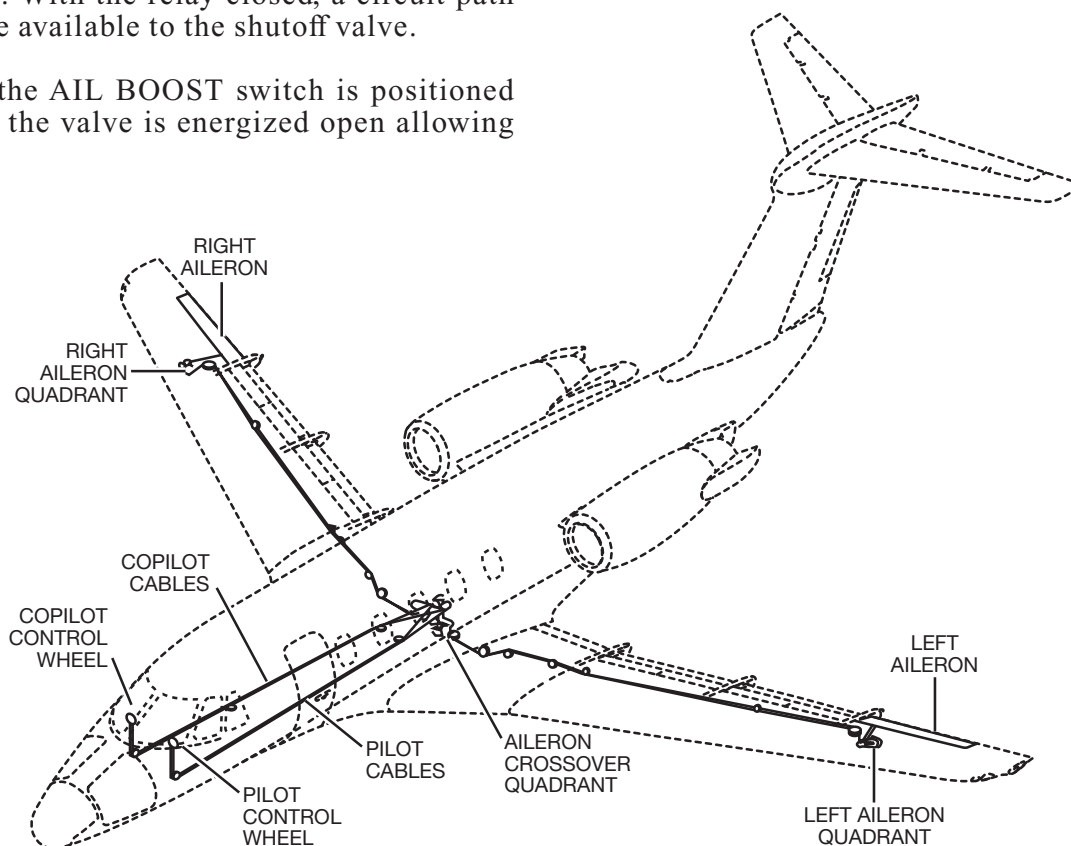


Figure 15-2. Aileron Control Cable Routing

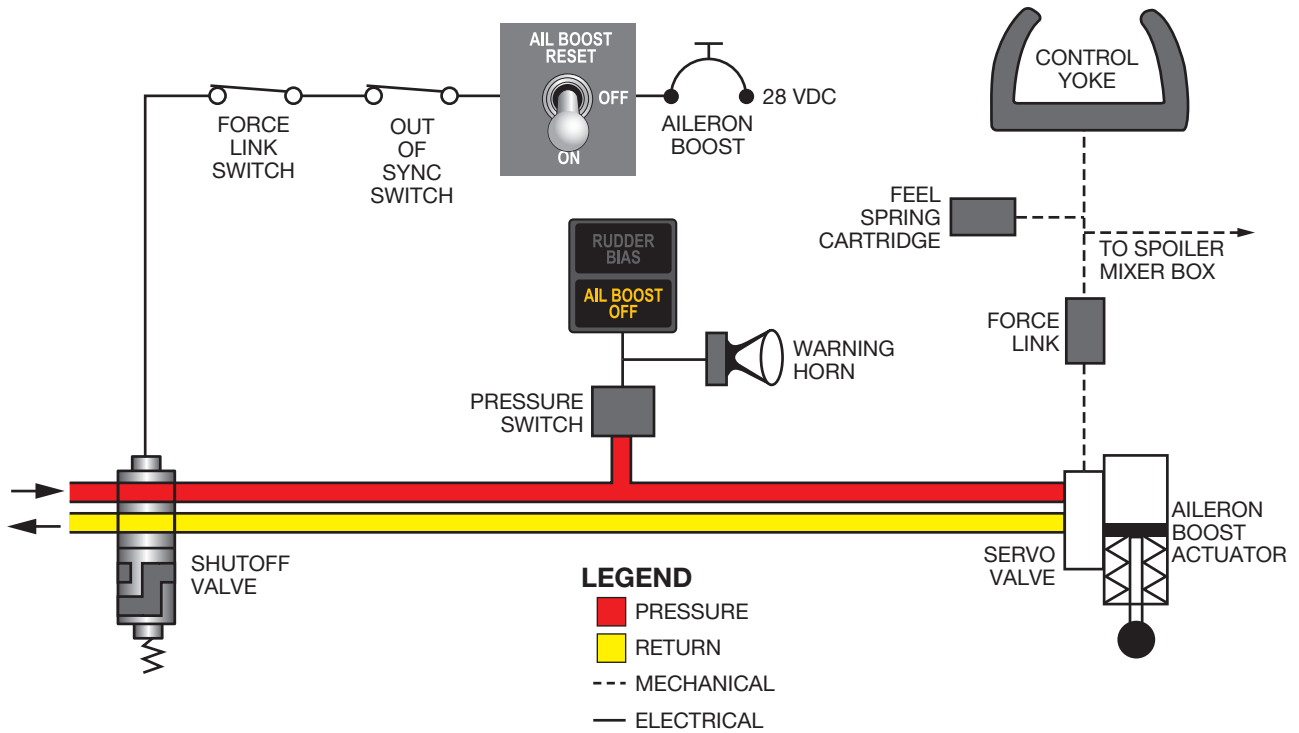


Figure 15-3. Aileron Boost System

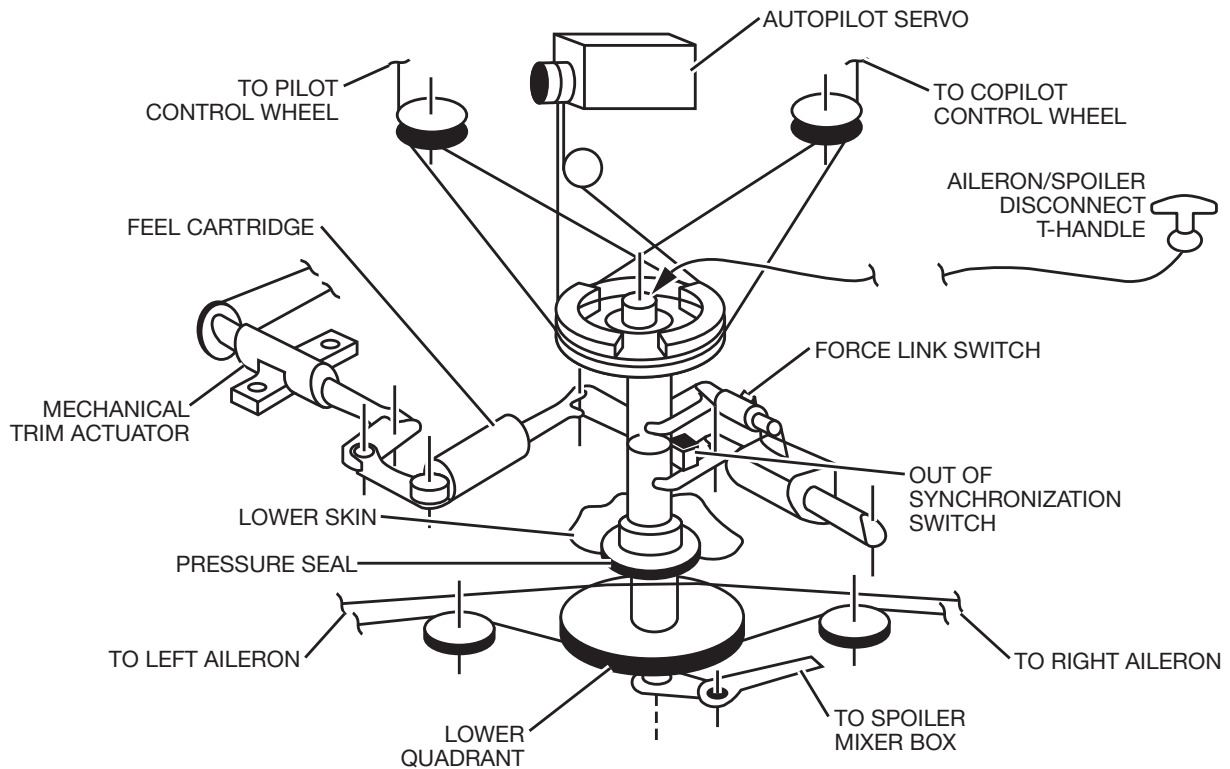


Figure 15-4. Aileron Crossover Quadrant



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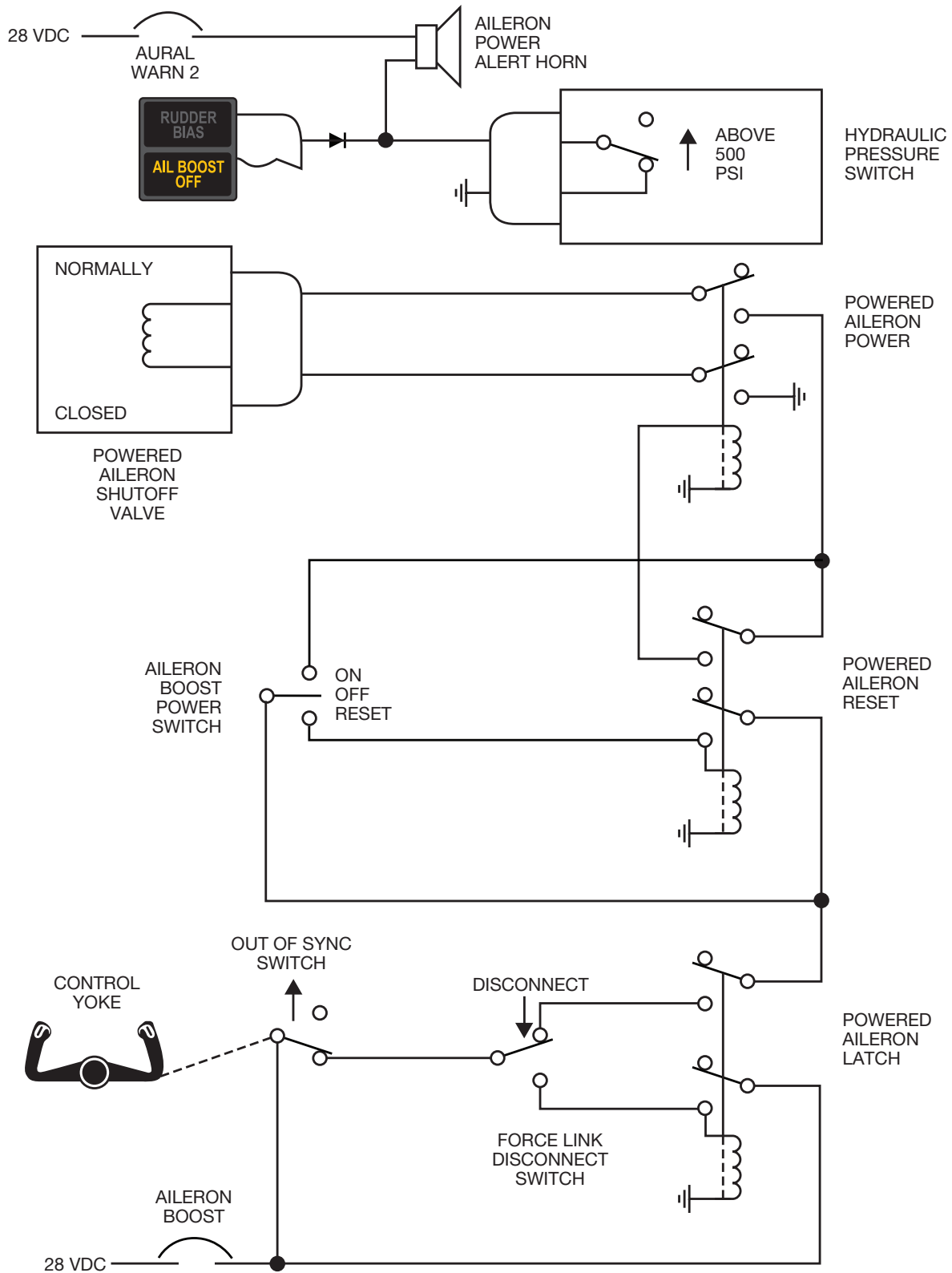


Figure 15-5. Aileron Power Schematic



DC power from the shutoff valve if the two shaft sections are not synchronized. This activates only if the force link is mechanically disconnected from the actuator, or if the linkage and the actuator movement did not follow the linkage. If the force link is disconnected upon initial power application, the valve cannot open because the force link must trip in order to enable circuit completion.

CONTROL LOCKS

Control locks, when engaged, lock the primary flight controls and prevent both throttles from being advanced more than 1.25 inches forward of IDLE. Prior to engaging the control locks, position both throttles near IDLE and neutralize the flight controls. Pull the CONTROL LOCK T-handle, located below the pilot instrument panel, and rotate it clockwise to lock the flight controls in neutral and both throttles near IDLE.

With the locks engaged, the amber GUST LOCK and NO TAKEOFF annunciators illuminate. To disengage the control locks, rotate the handle counterclockwise, and push it forward. The handle locks in the stowed position.

NOTE

The aircraft can be towed with the nosewheel deflected up to 90° with the control locks engaged.

WARNING

Do not tow or taxi the aircraft if the nose gear strut is extended beyond 7 inches.

SECONDARY FLIGHT CONTROLS

RUDDER TRIM

Rudder trim is mechanically operated by cables from a trim wheel on the pedestal. It is initiated by rotation of the rudder trim wheel on the

pedestal (Figure 15-6). Cables transmit motion to position the trim tab on the rudder. Trim position is indicated on a mechanical indicator adjacent to the trim wheel. The tab also functions as a servo tab, deflecting approximately half the angular travel of the rudder in the opposite direction to reduce the rudder force.

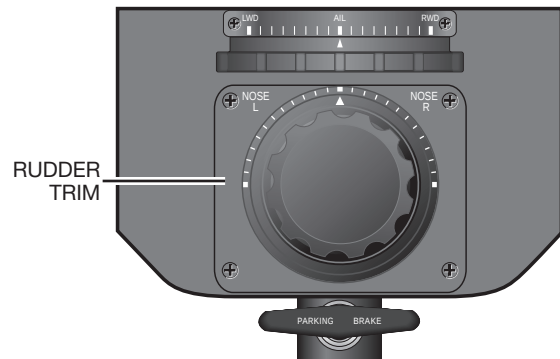


Figure 15-6. Rudder Trim Wheel

Rudder Bias System

The rudder bias system reduces yaw forces resulting from engine failure, especially at high power settings. At engine failure, the system applies the equivalent of 125–150 pounds of rudder pedal force on the side of the good engine. When on, the system is fully automatic in operation.

Operation

Positioning the RUD BIAS switch to NORM deenergizes the bypass valve closed, enabling the system (Figure 15-7).

If either engine loses power or fails, pressure to one end of the actuator is reduced accordingly, and the rudder deflects against the resulting asymmetrical thrust (Figure 15-8).

If electrical power is lost, the system continues normal operation, since the bypass valve is normally in the deenergized position.

Positioning the RUD BIAS switch to OFF (Figure 15-9) energizes the bypass valve open, which disables the system. Illumination of the



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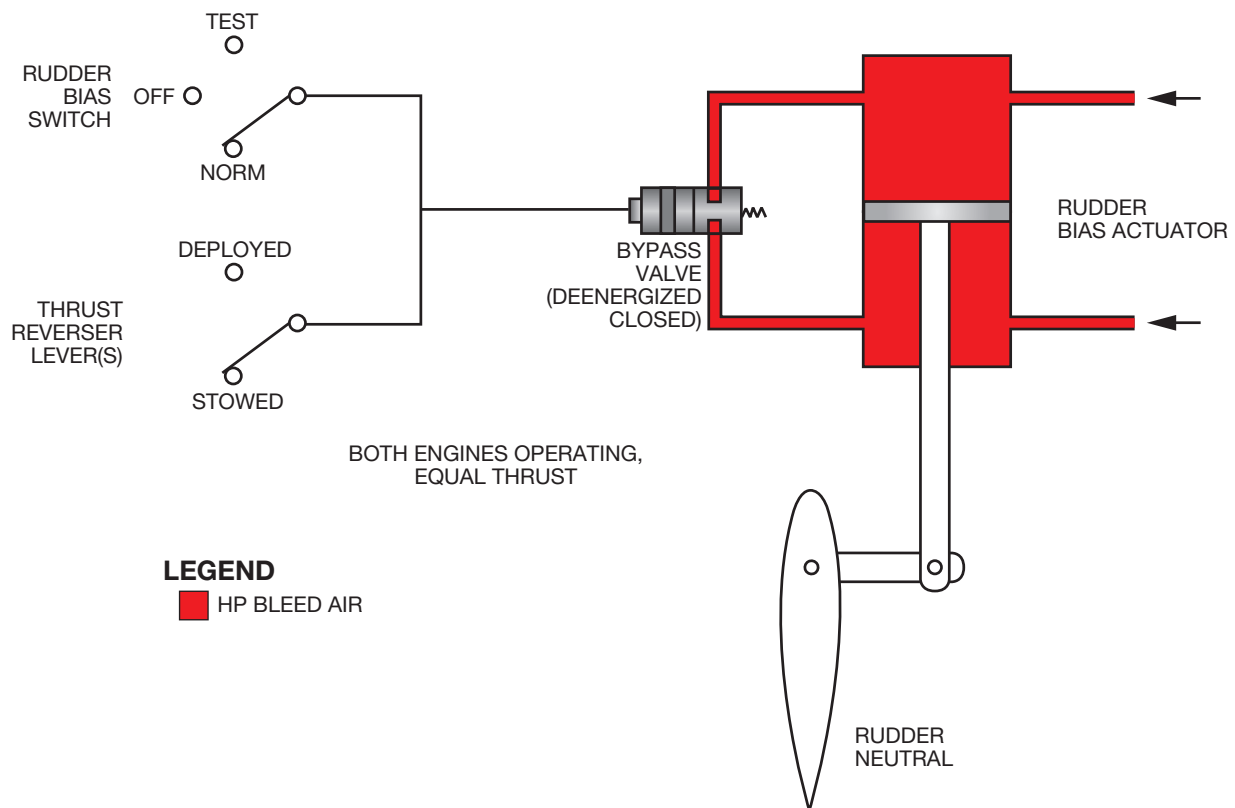


Figure 15-7. Rudder Bias System

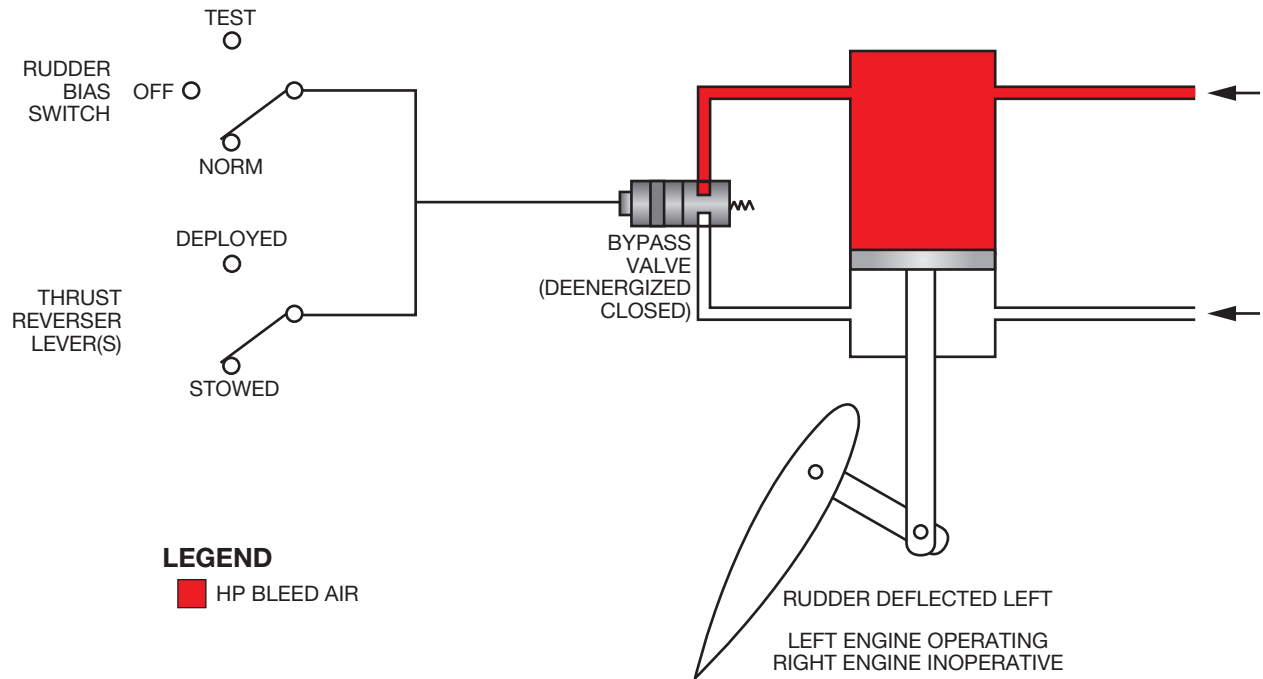


Figure 15-8. Rudder Bias Unequal Thrust

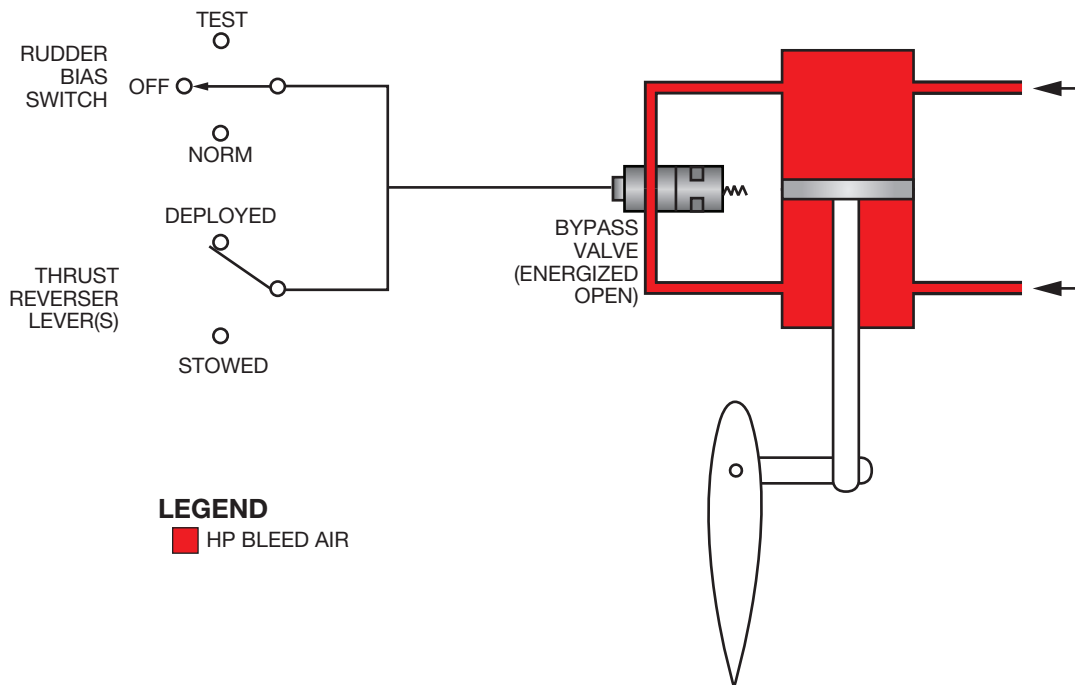
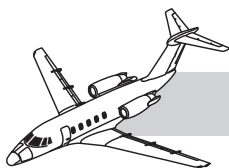


Figure 15-9. RUD BIAS Switch Off or TR Deployed



amber RUDDER BIAS annunciator indicates that the system is disabled.

Raising one or both thrust reverser levers to the deploy position also energizes the bypass valve open and illuminates the RUDDER BIAS annunciator. Before increasing the reverse thrust, verify that the RUDDER BIAS annunciator is illuminated.

The actuator is anti-iced by an electrical heat blanket powered through the RH PITOT/STATIC switch on the center panel. A thermal switch on the blanket cycles the power on and off.

Positioning the RUD BIAS switch to TEST applies power to the actuator heating element, bypassing the thermal switch. A minimum reading of 7 amps on the RUDDER BIAS HTR ammeter (Figure 15-7) indicates proper operation.

Uncommanded rudder movements on the ground with both engines operating constitute a system malfunction and must be corrected prior to flight.

AILERON TRIM

To initiate aileron trim, rotate the aileron trim wheel on the pedestal (Figure 15-10). Mechanical and cable linkage repositions the neutral point of the aileron feel cylinder for a slight aileron deflection. Rotation of the trim wheel relocates the aileron neutral position which is reflected on the control wheel position. An indicator on the pedestal shows the amount

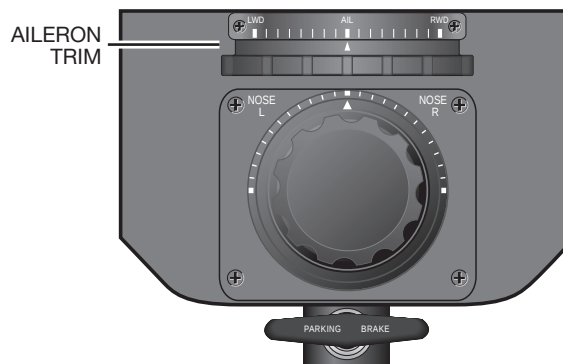


Figure 15-10. Aileron Trim Wheel

of trim displacement in relation to the original neutral position. Neither aileron has a tab.

PITCH TRIM

Two pitch trim systems, primary and secondary are available for each aircraft. The autopilot, when engaged, uses the primary trim for relief from maintaining constant force on the elevator servos.

Both pitch trim controls consist of split switches. For each switch, both halves must be moved simultaneously to achieve trim. Trim operation with one-half of the switch constitutes a malfunction (Figure 15-11).

Pitch trim is accomplished by moving the entire horizontal stabilizer. Two electric motors drive the leading edge of the stabilizer up or down.

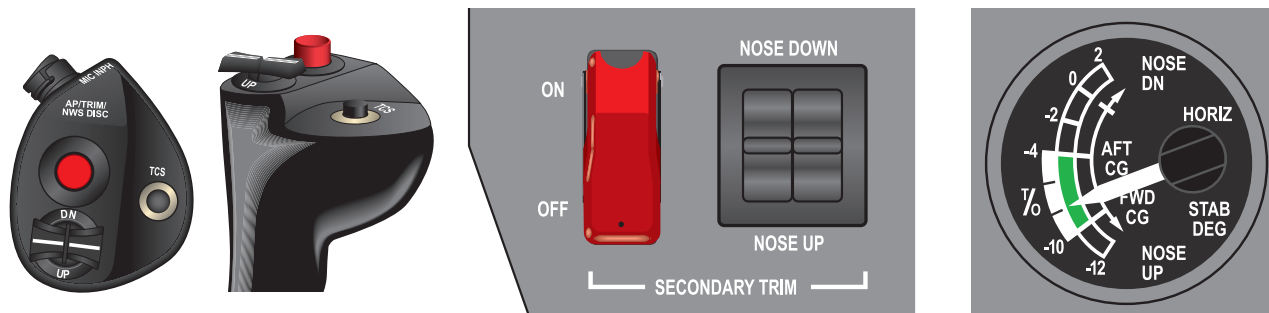


Figure 15-11. Pitch Trim and Indicators



Primary Pitch Trim

The primary trim system is powered by 28 VDC from the hot battery bus through the PITCH PWR and PITCH CONTROL CBs on the left cockpit CB panel.

The pitch trim system is operated with the split trim switches on the outboard side of either control wheel.

Positioning both halves of the switch to UP or DN applies power to the primary stabilizer trim motor, which positions the stabilizer leading edge as selected. Approximately 40 seconds is required for the stabilizer to travel from limit to limit of +2 to -12°. However, in normal operation, travel is restricted by limit switches.

The HORIZ STAB position indicator displays stabilizer travel, and the audio system transmits a clacker sound if the stabilizer is in motion for more than 1 second (Figure 15-11).

To disengage primary pitch trim, press the red AP/TRIM/NWS DISC button adjacent to the trim switch. Illumination of the red PRI TRIM FAIL annunciator indicates system malfunction (Figure 15-11).

Secondary Pitch Trim

The secondary trim system is powered by 28 VDC from the emergency branch bus, through the SEC PITCH CB on the left cockpit CB panel.

To arm secondary pitch trim (and simultaneously disarm primary pitch trim), position the red guarded SECONDARY TRIM switch on the pedestal to ON. Doing so illuminates the SEC TRIM FAULT annunciator (Figure 15-11).

Simultaneously moving both halves of the adjacent secondary trim switches to NOSE DOWN or NOSE UP extinguishes the SEC TRIM FAULT annunciator and actuates the secondary trim motor in the selected direction. If the amber SEC TRIM FAULT annunciator remains illuminated, the secondary trim clutch

did not engage. Approximately 74 seconds is required for stabilizer travel from one limit to the other.

The HORIZ STAB position indicator displays stabilizer travel, and the audio system transmits a clacker sound if the stabilizer is in motion for more than 1 second. To disengage the secondary pitch trim, position the guarded switch to OFF. With the system switched off, illumination of the annunciator indicates failure of the secondary trim clutch to disengage.

FLAPS

The six-segment Fowler flaps are actuated by a 28 VDC motor. Flexible drive shafts and mechanical actuators in the wing transmit motion to the flaps. Split flap protection is provided to disable the system if flap asymmetry occurs. A flap controller monitors and controls the system.

Operation

The flaps are controlled by the FLAP handle on the pedestal (Figure 15-12). The handle has stops (detents) at UP, 7°, 20°, and FULL positions.

Positioning the FLAP handle in one of the flap detents causes the flap controller to operate

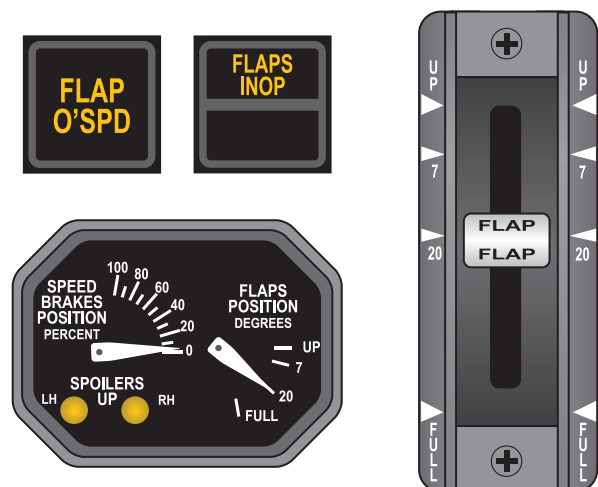


Figure 15-12. Flap Controls and Indicators



the flap motor. As the flaps move, position is shown in degrees on the FLAPS POSITION indicator on the pedestal. The flaps stop at the preselected detent position.

An amber FLAPS INOP switchlight (Figure 15-12) near the FLAP handle illuminates if the system malfunctions. If the system is disabled by the controller, the FLAPS INOP segment illuminates. Pressing the switchlight extinguishes it, but it illuminates again if the malfunction remains. If the flap handle is left in an intermediate position (between detents), the FLAPS INOP light illuminate and the flaps do not move. If this occurs, position the flap handle to a detent and press the FLAPS INOP switchlight to reset the system.

The FLAP O'SPD annunciators (Figure 15-12) if installed, are above the pilot and copilot altitude indicators. The annunciators illuminate if airspeed is above 215 KIAS and the flaps are extended beyond 1.2°.

If a split-flap condition occurs, the flap controller detects the condition, stops the flaps at the current position, and illuminates the FLAPS INOP switchlight.

To test the flap control system, position the rotary TEST knob to TRIM/FLAP. The FLAPS INOP switchlight illuminates, and then extinguishes if no faults are detected, and the FLAP O'SPD annunciators, if installed, illuminate.

Flaps Up Dispatch

The Citation VII has a no takeoff override system for disabling flap input to the NO TAKEOFF warning system when the flaps are up. An annunciator within the annunciator cluster above the electronic attitude direction indicator (EADIs) on each instrument panel, is used to activate the override system (Figure 15-13). The white, FLAPS UP portion of the annunciator illuminates if the aircraft is on the ground with the flaps up. When the switchlight is pushed on the pilot side only, the lower half of both switchlights illuminates the green FLP-NTO O'RIDE portion of the annunciator, indicating that flap input is disabled. All other

inputs to the NO TAKEOFF warning system are unaffected.

NOTE

Refer to the *Aircraft Flight Manual* for limitations, operating procedures, and performance data for dispatch with flaps in the up position.



Figure 15-13. FLAPS UP and FLP-NTO O'RIDE Annunciator

SPOILERS

The aircraft have eight manually controlled and hydraulically actuated spoiler panels on top of the wing. The outboard spoiler panel is used to assist the aileron in roll control. The two center panels on each wing are used as speedbrakes. The inboard panel is normally used after landing and is referred to as the ground spoiler.

Roll Spoilers

Control wheel movement for aileron operation also provides a mechanical input to the spoiler mixer box which sends a mechanical output via cables to operate the servo valve on the roll spoiler actuator on the down wing. The mixer box does not signal roll spoiler deflection until the aileron travels approximately 3.5° up. Normally, only one roll spoiler is deflected up at a time. However, actuation of the spoiler lever on the speedbrake lever (Figure 15-14) positions both roll spoilers to full up, in which case the roll spoilers no longer respond to control wheel roll inputs.

The both full up roll spoiler condition is normally accomplished after landing; in flight, the both full up position is used for emergency descents and in some cases for a jammed stabilizer.

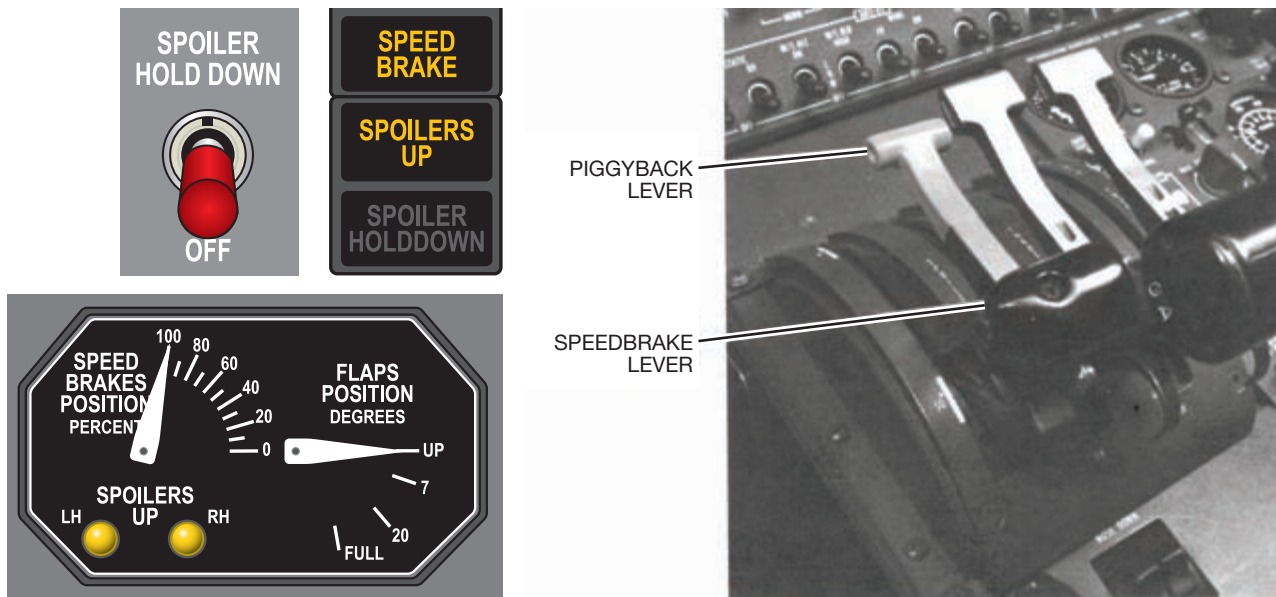


Figure 15-14. Speedbrakes and Spoilers Up

Speedbrakes

The speedbrakes consist of the two center spoiler panels on each wing. They can be hydraulically deployed to any position up to 50° by selective positioning with the speedbrake lever (Figure 15-14).

Moving the speedbrake lever aft simultaneously positions all four speedbrakes in proportion to lever movement. The **SPEED BRAKES** indicator shows speedbrake position. With the lever full aft, the speedbrakes are at 50° (100%) extension. Whenever the speedbrakes are not fully stowed, an amber **SPEED BRAKE** annunciator illuminates.

The use of speedbrakes is prohibited below 500 feet AGL with the flaps out of the **UP** position. In flight, a flap/speedbrake monitor system sounds the no-takeoff warning horn if the flaps are not up, the speedbrakes are deployed, and the radio altimeter altitude is indicating below 500 feet AGL.

Ground Spoilers

The inboard spoiler panels are used as ground spoilers. Unlike the roll spoilers and speed-

brakes, the ground spoiler panels can be commanded to two positions only—full up (30°) and stowed. Also unlike the others, the signal to command full up deployment is hydraulic to the servo valve on the actuator.

Main system hydraulic pressure is directed to the servo by a spoiler control valve, which is actuated by the spoiler piggyback lever on the speedbrake lever (Figure 15-14).

The ground spoilers are individually monitored by amber lights inside the speedbrake position indicator. These illuminate when the left and right panels are not fully stowed. If either of these lights illuminate, an amber **SPOILERS UP** annunciator illuminates as well.

Emergency Descent and Ground Spoiler Operation

To configure the spoilers for emergency descent, fully extend the speedbrakes, and then raise the piggyback lever on the speedbrake lever. Doing so sends main system hydraulic pressure to the spoiler mixer box and the ground spoiler servo valves, causing both roll spoilers and both ground spoilers to fully deploy. In this configuration, the roll



spoilers do not operate with the ailerons, and roll authority is reduced.

Spoiler Hold Down

The spoilers are held in the stowed position by hydraulic pressure commanded by the servo valves. Additionally, the spoiler actuators respond only to down commands if the spoiler system pressure drops below 1,200 psi.

If system pressure is lost, the spoilers can float up causing an increase in drag with a resulting decrease in performance. To guard against this, a hold down system is incorporated. It is important to understand that the only conditions that cause spoiler hold down to activate are loss of hydraulic pressure or the pilot actuating the SPOILER HOLD DOWN switch in the cockpit (Figure 15-14).

Spoiler Operation

Main system pressure enters the spoiler module and passes through the spoiler hold down valve. The pressure normally takes two paths. Figures 15-15 through 15-20 illustrate the different conditions for spoiler operation.

Pressure to the spoiler actuators also operates a mechanical valve which is held in a position that prevents pressure from the spoiler hold down accumulator from entering the hold down line. The other path taken is to the spoiler hold down accumulator. It is charged by main system pressure and has a nitrogen precharge of 1,500 psi.

If the SPOILER HOLD DOWN switch is placed in the HOLD DOWN position, DC power is applied to energize the spoiler hold down module to the position shown in Figure 15-18. Pressure to the servo valves on the actuators and to the mechanical dump valve on the spoiler hold down line is shutoff. The path from the hold down accumulator to the stow side of the spoiler actuators is opened by the dump valve, and the spoilers stow. However, since main hydraulic pressure is still available to the spoiler hold down accumulator, it is actually the main system pressure holding

the spoilers down. The effect is the same if main system pressure is lost with the spoiler hold down valve in its deenergized position. Pressure is lost to the servo valves, and the dump valve opens to allow the stored accumulator pressure to stow the spoilers.

In addition to main system pressure charging the hold down accumulator, the electric auxiliary pump can also charge the accumulator through a check valve. The AUX HYD PWR switch, located on the center switch panel, is a three position switch labeled NORM-OFF-ON (Figure 15-15).

With the switch in the ON position, the auxiliary pump is powered on and supplies pressure to the wheel brakes, the spoiler hold down accumulator, two pressure-operated valves on the hold down lines to the roll spoiler actuators, and, through the energized-open auxiliary pressure dump valve, to the servo valve on the roll spoiler actuators. This allows the auxiliary pump to hold the speedbrakes and ground spoilers down and to restore operation of the roll spoilers to assist in lateral control.

With the AUX HYD PWR switch in OFF, the pump and the auxiliary dump valve are both deenergized.

If the switch is in NORM, the auxiliary pump activates automatically under two conditions: the main system pressure dropping below 1,200 psi, or the SPOILER HOLD DOWN switch being placed in the HOLD DOWN position. If operation of the pump is not desired, the AUX HYD PWR switch can be placed in the OFF position, and the pump ceases operation.

CAUTION

Ensure that the AUX HYD PWR switch is OFF when the aircraft normal DC power is removed after flight. Otherwise, the pump, which is powered from the hot battery bus, drains the aircraft batteries.

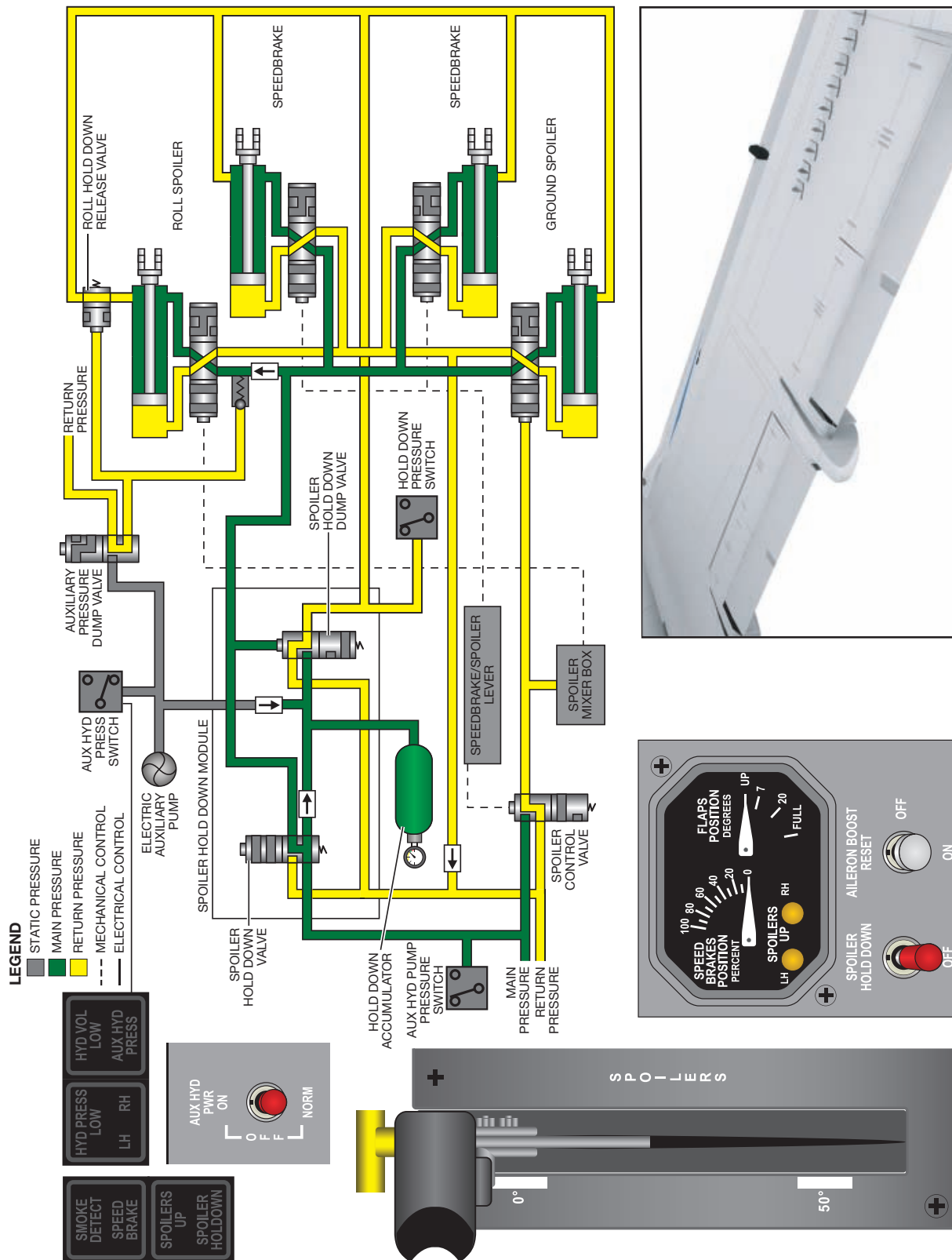
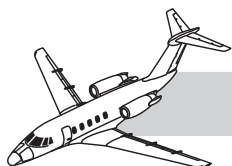


Figure 15-15. Spoilers Normal Operation

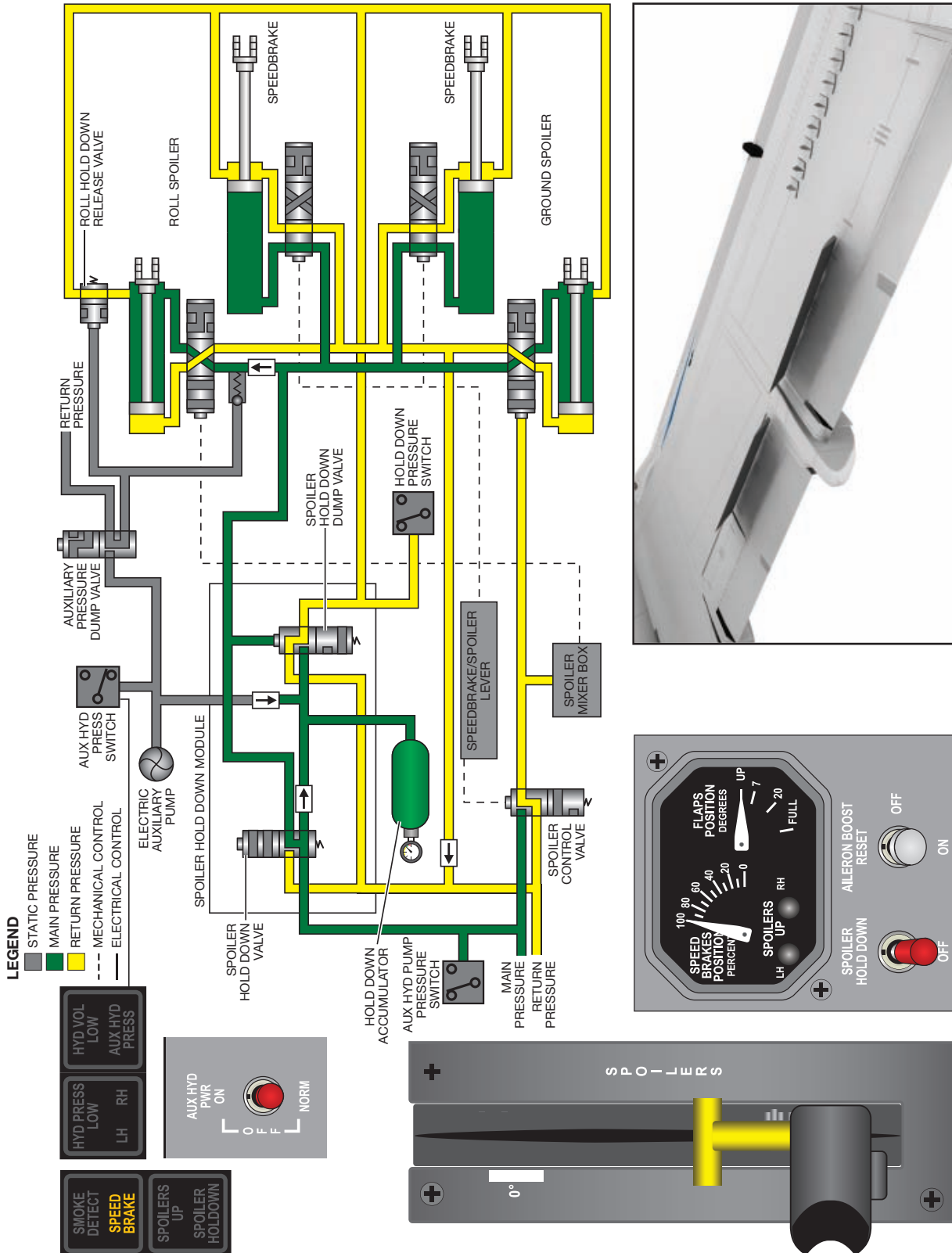


Figure 15-16. Speedbrakes Up

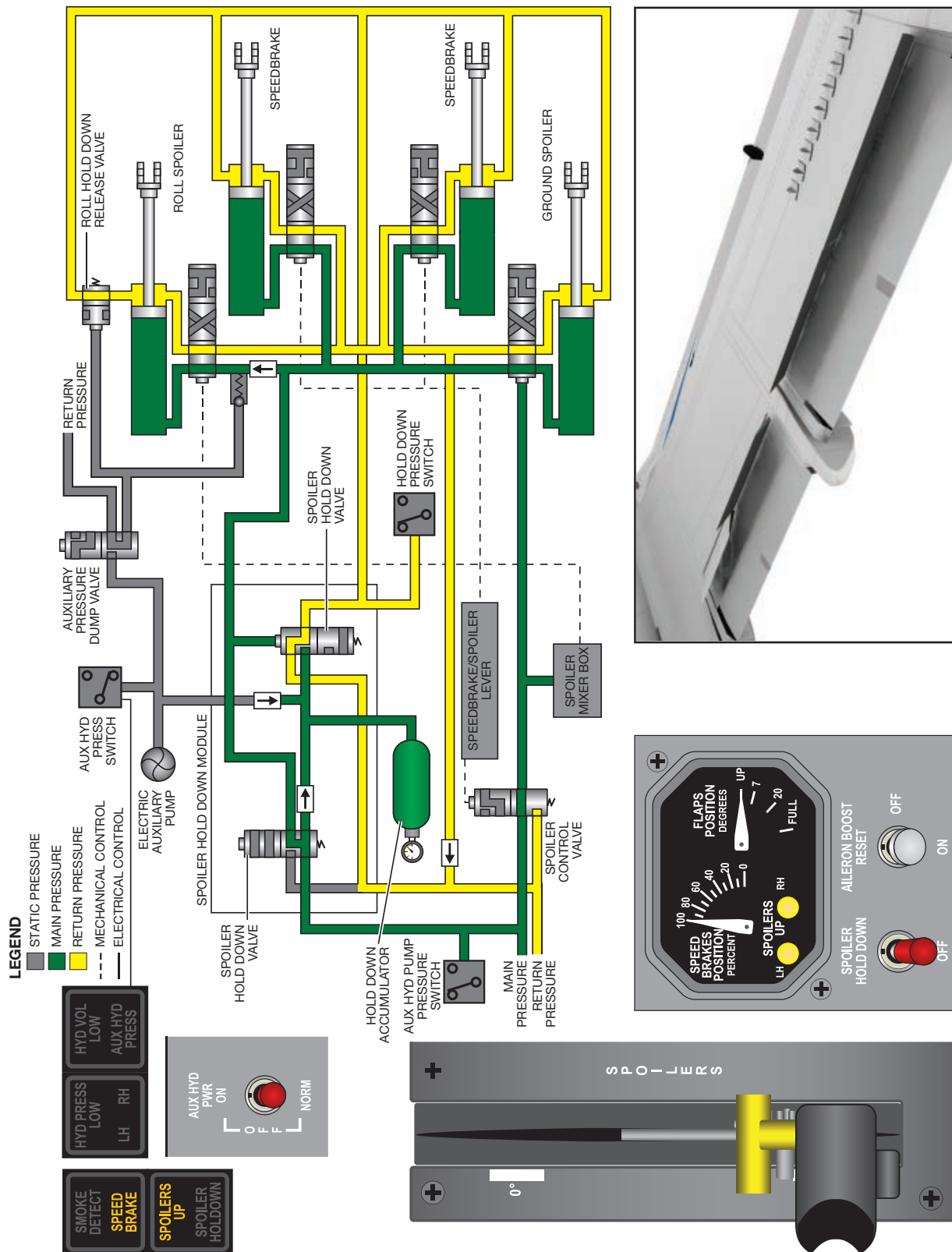
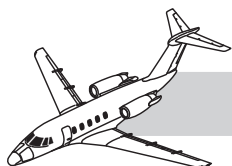


Figure 15-17. Spoilers Up

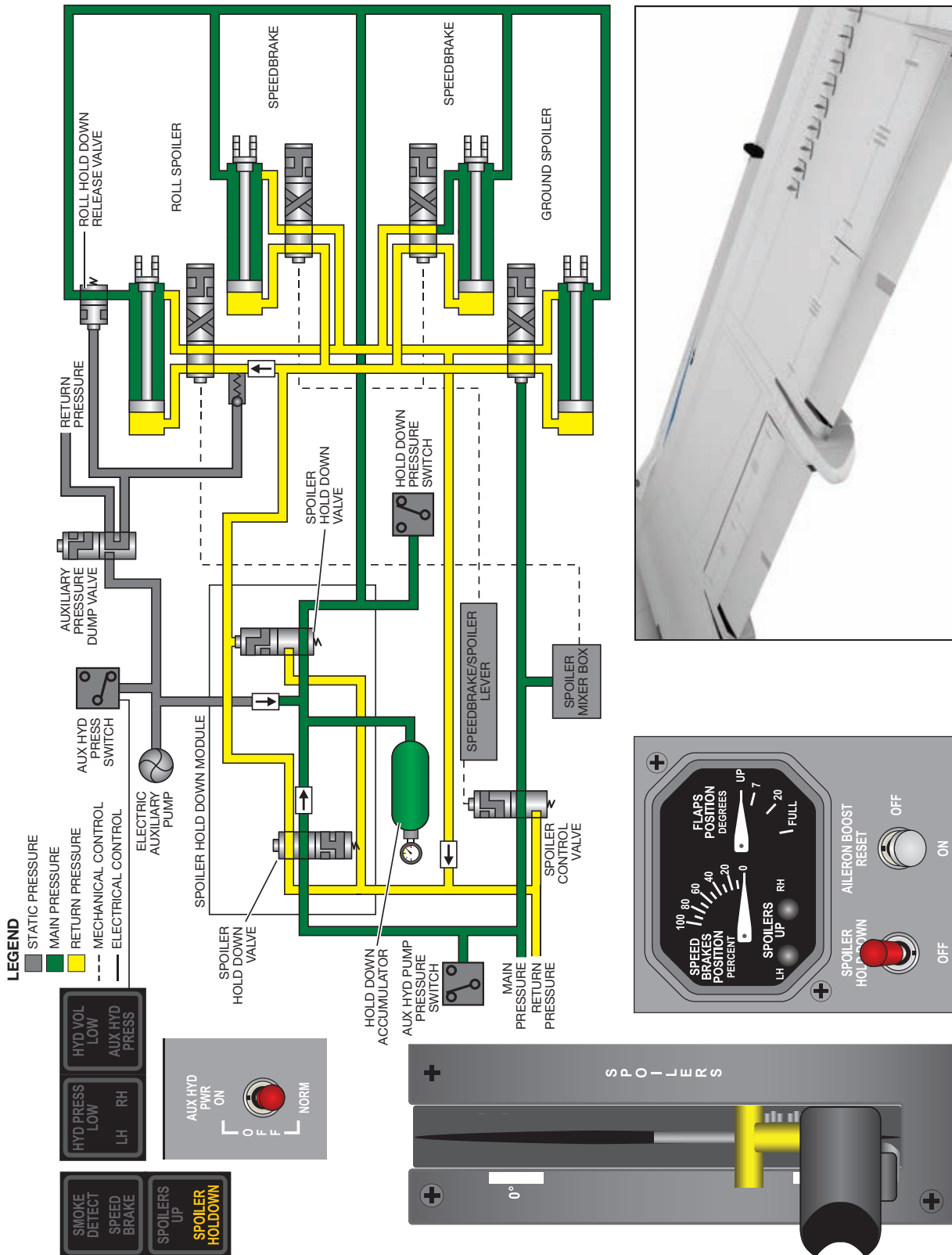


Figure 15-18. Spoiler Hold Down Selected—AUX HYD PWR Switch OFF







If both main and auxiliary hydraulic pressure are lost, the spoiler hold down accumulator should continue to provide sufficient pressure to keep the spoilers fully stowed.

Hydraulic Fluid Temperature Monitoring System

The speedbrake and roll spoiler actuators have thermostatic valves, which open to circulate warm fluid from the main system into the actuators when the actuator fluid temperature drops to -10°F (-23°C). The system prevents actuator fluid from becoming cold soaked and ensures that speedbrakes and/or ground spoilers do not deploy sluggishly, causing an uncommanded roll.

The hydraulic temperature monitoring system monitors fluid temperature at the speedbrake and roll spoiler actuators. It also monitors fluid temperature in the return line from the spoiler system. If the temperature at any of the eight sensors drops to -27°F (-32°C), then two HYD TEMP LOW annunciators are illuminated. The annunciators are near the top of the left and right instrument panels (Figure 15-21). If the annunciators illuminate in flight and cannot be extinguished using the checklist procedures, then speedbrake and ground spoiler use is prohibited and the aircraft must not exceed 45,000 feet. If the annunciator subsequently extinguishes, then the restrictions are removed.

The hydraulic temperature monitor box is in the tailcone and has built-in test equipment

(BITE) (Figure 15-22). The monitor box includes a red flag indicator, visible through a window on the box cover, which indicates which sensors activated the HYD TEMP LOW annunciators. The red flag remains in place even if the annunciators extinguish. The BITE indicator can be reset by the crew or maintenance.



Figure 15-22. Hydraulic Temperature Monitor Box

AILERON/SPOILER DISCONNECT

Since there are two separate roll control systems, certification regulations require that a means be provided to separate them. This prevents a jam or failure in one system from disabling the entire system.

The red AILERON–SPOILER DISCONNECT T-handle (Figure 15-23) is used to disconnect

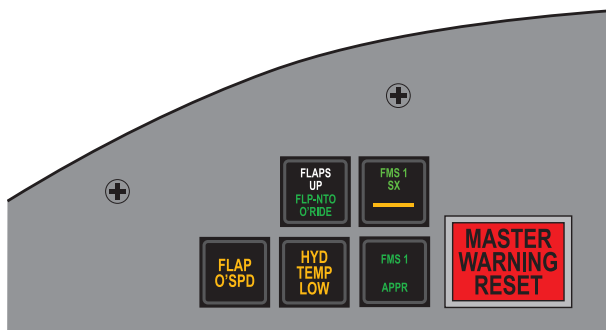


Figure 15-21. HYD TEMP LOW Annunciator



Figure 15-23. AILERON–SPOILER DISCONNECT T-Handle



the two systems. The handle is on the pedestal and covered with a clear plastic guard.

If a jam occurs in lateral control, pull the T-handle. Doing so separates the quadrants, which are normally pinned together (Figure 15-24). The pilot wheel controls the ailerons, which still have load feel and the copilot wheel controls the roll spoilers which have no load feel. Once the handle is pulled, the quadrants cannot be reconnected in flight.

NOTE

This system is to be used only under a jammed roll control condition. It is not to be used for any other purpose.



Figure 15-24. Aileron/Spoiler Disconnect

ANGLE-OF-ATTACK AND STALL WARNING SYSTEM

GENERAL

An indicator, a speed command on the ADI and an indexer provide angle-of-attack (AOA) readings. A stick shaker on each control column provides stall warning. The AOA probes, flap controller, and spoiler and speedbrake position sensors provide inputs to the system. The right branch DC bus supplies electrical power.

OPERATION

The AOA probes, one on each side of the forward fuselage, detect airflow angle and provide signals to the AOA computer. The computer also receives AOA information from the flap controller, and spoiler and speed-brake position sensors. All of the inputs drive the pointer on the face of the indicator.

The indicator has four colored arcs: green, white, yellow, and red (Figure 15-25). The green arc (0–0.55) is the normal operating range of the aircraft. The white arc (0.55–0.65) covers the area between the normal operating range and the caution area. The middle of the white arc (0.6) represents the optimum landing approach airspeed (V_{APP} or V_{REF}).

The yellow range (0.65–0.75) is a caution area where the aircraft is approaching a critical AOA. The red arc (0.75–1.0) is a warning area and represents the beginning of low speed buffet to full stall. At an indication of $0.82 \pm .02$ in the warning range, the stick shakers are activated.

The indicator displays lift information with 0 representing zero lift and 1.0 representing 100% lift capability. The lift is presented as a percentage and with speedbrake and flap position information, the display is valid for all aircraft configurations and weights. Therefore, at 1.0 where full stall occurs, 100% of available lift is being produced.

The approach indexer shows a heads up display of deviation from the approach reference (Figure 15-25). The display is represented by three lighted symbols, which indicate relative speed to the approach reference. The following AOA indications occur:

- AOA high—top (red) chevron lighted
- AOA on reference—(green) circle lighted
- AOA low—bottom (yellow) chevron lighted



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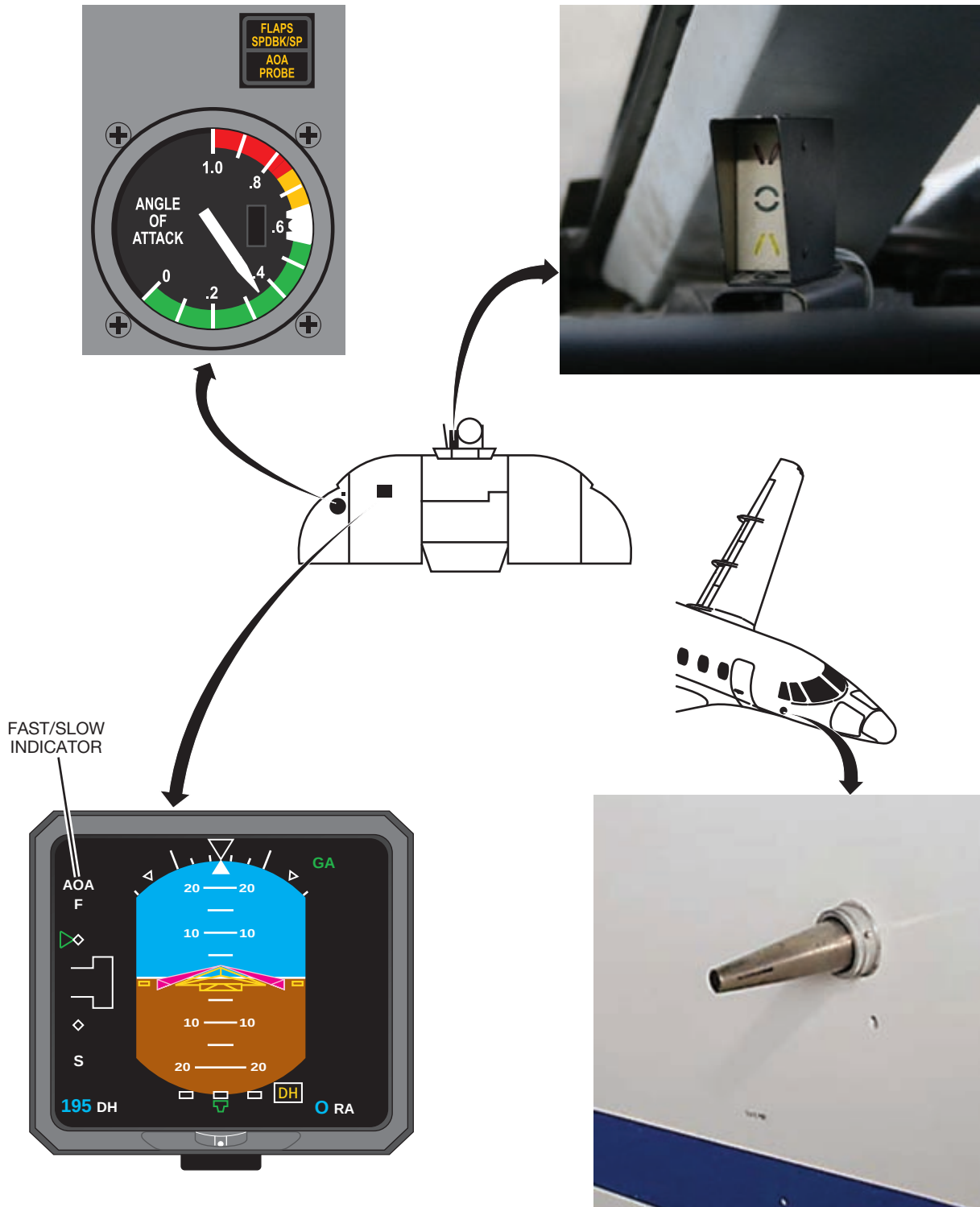


Figure 15-25. Angle-of-Attack System Components



The top chevron points down, indicating that the pitch attitude must be decreased. The bottom chevron points up, indicating that the angle of attack can be increased. A second, optional indexer for the copilot is available.

Two amber annunciators are above the ANGLE OF ATTACK indicator (Figure 15-25). Illumination of the AOA PROBE annunciator indicates a fault in a probe or transmitter, but the system remains operative since the computer selects the most conservative input.

Faulty inputs from the flaps or speedbrake sensors illuminate the FLAPS SPDBK/SP annunciator, and the system selects a stick shaker actuation range of 0.715 to 0.825 in the red arc. The OFF flag is visible and the indexer and the FAST/SLOW indicators on the attitude director indicators (ADIs) are inoperative. If the ground spoilers are deployed, the same indications are displayed.

If a fault occurs in the stall warning system that renders it inoperative, the amber STALL WARN annunciator illuminates. The ANGLE OF ATTACK indicator OFF flag is visible, the FAST/SLOW indicators on the ADIs are inoperative, and the stick shakers do not operate.

LIMITATIONS

For specific information on limitations, refer to the appropriate abbreviated checklists or the FAA-approved *Airplane Flight Manual (AFM)*.

EMERGENCY/ ABNORMAL

For specific information on emergency/abnormal procedures, refer to the appropriate abbreviated checklists or the FAA-approved *AFM*.



QUESTIONS

1. The manually actuated primary flight controls are:
 - A. Rudder and elevators
 - B. Rudder only
 - C. Ailerons and horizontal stabilizer
 - D. Elevators only
2. The roll control spoilers:
 - A. Cannot be operated without powered ailerons
 - B. Can be operated with the hold down switch in HOLD DOWN
 - C. Cannot be operated without aileron movement
 - D. Cannot be operated with manual ailerons
3. Control feel in the aileron system is provided by:
 - A. A hydraulic damper
 - B. A spring cartridge
 - C. Airload only
 - D. Friction in the control system
4. Aileron boost can be disabled by:
 - A. DC power failure
 - B. The force link switch
 - C. The out of synchronization switch
 - D. All of the above
5. If the AILERON SPOILER DISCONNECT T-handle is pulled:
 - A. Aileron trim is available
 - B. The right control wheel is connected to the roll spoilers and the feel cylinder
 - C. The roll spoilers cannot be extended by the spoiler lever
 - D. Only manual ailerons are available
6. Regarding the gust lock:
 - A. The engines cannot be started with it engaged
 - B. The aircraft should not be towed with it engaged
 - C. Turning the nosewheel greater than 60° with the gust lock engaged damages the nosewheel steering
 - D. It makes no difference if it is engaged for towing
7. The rudder bias system:
 - A. Provides power boosting for the rudder
 - B. Applies the equivalent of 125–150 pounds of rudder force on the side of the good engine
 - C. Applies the equivalent of 125–150 pounds of rudder force on the side of the failed engine
 - D. Locks the rudder in neutral if an engine fails
8. Concerning pitch trim:
 - A. Primary trim uses avionics AC power
 - B. Secondary trim operates slower than primary trim
 - C. With both inverters inoperative, primary trim must be used
 - D. The autopilot trims through the primary system at normal speed
9. The true statement regarding the flight control is:
 - A. If one flap power drive unit fails, a split flap condition can occur
 - B. Rudder trim is accomplished electrically from the rheostat knob in the cockpit
 - C. The autopilot trims via the secondary pitch trim system while the pilot trims via the primary
 - D. If the flap handle is positioned other than in a detent, the FLAPS INOP annunciator illuminates



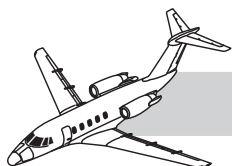
10. An incorrect statement concerning the rudder system is:
- A. Before thrust reverser deployment, the RUD BIAS switch must be positioned to OFF
 - B. The rudder bias system is pneumatically operated
 - C. The rudder trim tab is manually operated
 - D. The rudder trim tab has a servo action
11. The wing flaps:
- A. Can only be selected to flap selector detent positions
 - B. Cannot be reset if the controller overheats
 - C. Lock in position before a serious split flap condition can occur
 - D. Require 6 seconds to fully retract from the 37° position
12. The inboard (ground) spoilers:
- A. Stow if normal hydraulic pressure is lost
 - B. Always extend with the speedbrakes
 - C. Always extend when the spoiler lever is raised
 - D. Are controlled by the spoiler mixer box
13. The stall warning consists of:
- A. Only stall strips on the wing leading edges
 - B. A stall warning annunciator on the annunciator panel
 - C. Aerodynamic buffet and stick shakers
 - D. Stick shakers
14. The pitch trim setting for takeoff is determined by:
- A. Gross takeoff weight
 - B. Takeoff Weight Center of Gravity
 - C. Fuel loading
 - D. Pressure altitude and temperature



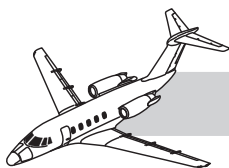
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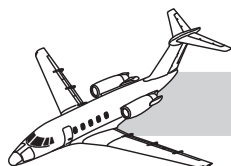
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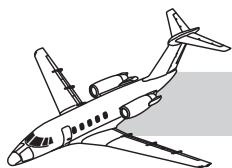


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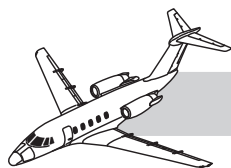


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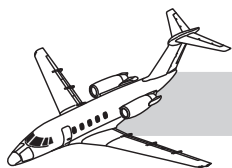
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INTRODUCTION

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FLIGHT GUIDANCE SYSTEM: SNs 7001—7119

GENERAL

The SPZ-8000 Digital Integrated Flight Controls System (DIFCS) consists of two independent Electronic Flight Instrument Systems (EFIS). One independently controlled system

on each of the pilot and copilot instrument panel (Figure 16-1). Each system (pilot and copilot) may be used in conjunction with the single flight guidance controller and the single



Figure 16-1. SPZ-8000 DIFCS Flight Deck: SNs 7001 – 7119

remote instrument controller to fly the aircraft. The crew can use the system to fly the aircraft manually using the flight director, or automatically using the Flight Guidance controller. The system meets category II equipment requirements when flown with the autopilot engaged.

The DIFCS includes the DFZ-800 Flight Guidance System (FGS), EDZ-816 Electronic Flight Instrument System (EFIS), AHZ-600 Attitude and Heading Reference System (AHRS), and the ADZ-810 Air Data System.

The SPZ-8000 DIFCS is a complete automatic flight control system. Flight path commands are generated by the FZ-800 Flight Guidance Computers which integrates attitude and heading from the AHRS, air data from the ADZ-810 system, navigation from various navigation components, and the EFIS all via an Avionics Standard Communications Bus (ASCB).

The DIFCS requires primarily 28 VDC power for operation. The RMIs, altimeters, and air-speed displays require 26 VAC. The weather radar system also requires 26 VAC for antenna stabilization.

DFZ-800 FLIGHT GUIDANCE SYSTEM

The DFZ-800 Flight Guidance System consists of an integrated autopilot/flight director, dual A-channel/B-channel Flight Guidance computers, and three flight control servos (elevator, aileron and rudder). Dual AHRS and air data computers provide fail-safe redundancy along with various sensor fail-safe characteristics. A single GC-810 Flight Guidance Controller is used to engage the autopilot and select operating modes for the flight director (Figure 16-2).

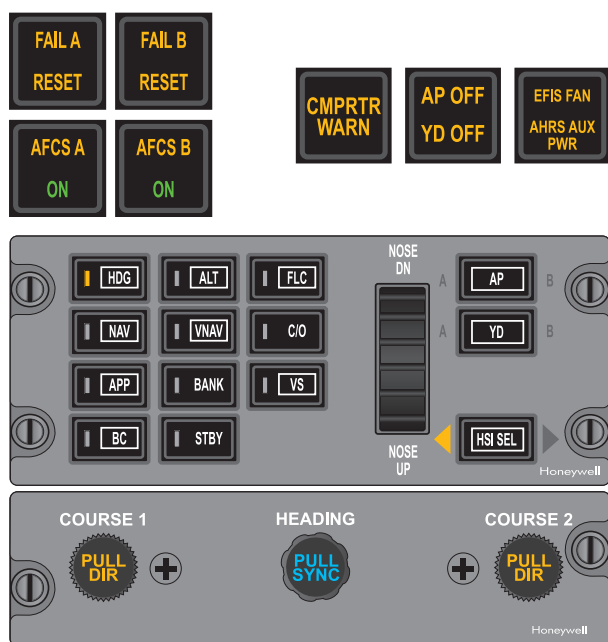
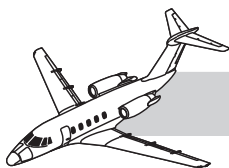


Figure 16-2. Flight Guidance and Instrument Remote Controllers and Associated Annunciators

Power-Up Test

Upon initial application of DC power, the Flight Guidance system initiates a self-test for approximately 30 seconds. If the test is satisfactory, the AFCS A instrument panel annunciator located on the pilot lower instrument panel will illuminate (Figure 16-2). The A-channel lights adjacent to the left side of the AP and YD engage switches will also illuminate (Figure 16-2). If a failure is detected in the A-channel Flight Guidance computer, the AFCS B instrument panel annunciator and the B-channel lights on the controller (Figure 16-2) will both illuminate.

If a failure is detected in either Flight Guidance computer, the appropriate FAIL A(B) RESET switchlight on the center instrument panel latches ON. The failed computer may be reset by pushing and releasing the appropriate FAIL A or B RESET switchlight. If the reset is satisfactory, the appropriate FAIL A or B annunciator will extinguish in approximately 2 seconds.

If a failure occurs within a Flight Guidance computer that disables the flight director func-

tions, the FD FAIL annunciator on the EADI will display. This malfunction will also cause the flight director command bar on the EADI to disappear.

If both Flight Guidance computers should fail, the second computer failure will disengage the DFZ-800 Flight Guidance System and the autopilot. The AP OFF annunciators above the EADIs will illuminate and the autopilot warning horn will sound.

Autopilot Functions

The autopilot and yaw damper switches on the controller provide for engagement and disengagement of the autopilot and yaw damper functions. They also provide manual control of the autopilot is through the PITCH wheel and the HEADING knob on the Instrument Controller (Figure 16-2). When engaged, the AP and YD switches will be illuminated and may be disengaged by various methods. To simultaneously disconnect the AP and YD, push the red AP/TRIM/NWS DISC button on the control wheels. The autopilot can also be disengaged as follows:

- Pressing the go-around button on the throttles (providing the flight director is engaged)
- Selecting the AOA/flap test with the rotary test switch
- Activating Primary Trim (doesn't disengage the yaw damper)
- Activating the AHRS manual-slave LH–RH switch
- Activating the AHRS fast-erect switch

NOTE

The autopilot cannot be engaged on the ground.

HSI Select

The HSI SEL button (Figure 16-2) is used to select either the pilot or copilot EHSI and DADC data for lateral and vertical guidance to both Flight Guidance computers. The arrow



lights adjacent to the selector switch displays which side is controlling the Flight Guidance computers. During an ILS approach, captured inbound, both Navigation receivers tuned to the same ILS frequency, and the aircraft within 1,200 feet of the terrain, both arrows illuminate. If one ILS receiver should fail, the arrow on that side extinguishes, the other arrow will remain illuminated, and the system will automatically select the remaining operating side for display.

When the HSI select button is used to transfer to the cross-side EHSD and DADC, all flight director modes are canceled. The Flight Director must be reprogrammed for the new side for operation.

NOTE

The arm and/or capture status of the Flight Director is only annunciated on the EADI.

Flight Director Modes

The Flight Director modes consist of the following:

Heading (HDG)

HDG annunciated in green along the top of the EADI (Figure 16-3), commands the Flight Guidance computer to track inputs of the heading bug on the EHSI (Figure 16-3). The heading bug is controlled by the HEADING knob on the Instrument Controller (Figure 16-2). While in this mode, a lower angle of bank limit (17°) may be selected with the BANK button (Figure 16-2).

Navigation (NAV)

VOR, LOC, AZ, LNAV are annunciated along the top of the EADI (white-arm, green-capture). The Flight Guidance computer is programmed to arm, capture, and track the selected navigation signals from the navigation receivers.

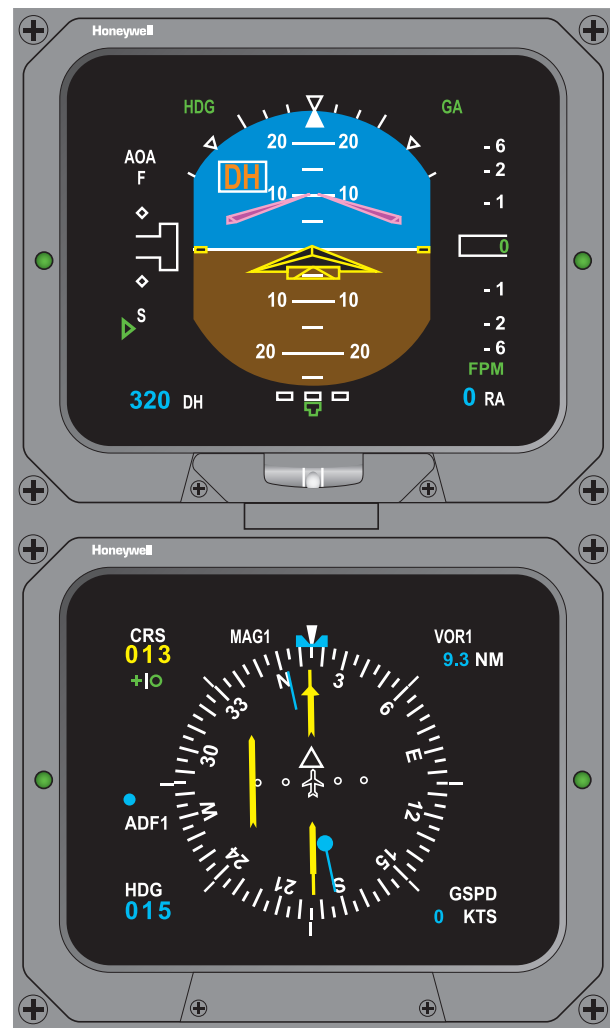


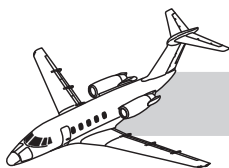
Figure 16-3. Electronic Flight Instrument System

Approach (APP)

Increases the signal gain to arm and capture lateral deviation signals for VOR, LOC and AZ and both lateral and deviation signals for ILS and MLS. The EADI will annunciate VAPP with a VOR frequency tuned.

Back Course (BC)

Commands the Flight Guidance computer to track the localizer back course. Reverses the directional input for the command bar.



NOTE

Ensure the inbound FRONT course is set in the course window on the EHSI. Set by the COURSE 1/2 knobs on the instrument remote controller.

BC displayed in white or green as appropriate along the top of the EADI.

Altitude (ALT)

Commands the system to hold the current aircraft altitude at the time of engagement. ALT is annunciated in green along the top of the EADI. Capturing the altitude displayed on the AL-801, the Altitude Preselect Controller will allow the system to maintain the displayed altitude (Figure 16-4).

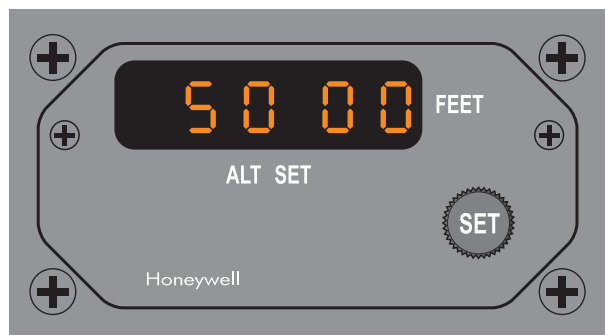


Figure 16-4. Altitude Preselect Controller

Vertical Navigation (VNAV)

Provides vertical path guidance from a compatible long range navigation system (Honeywell FMS only).

Bank (BANK)

Engaging this mode commands the Flight Guidance Computer to reduce the maximum bank angle when in the HDG mode (17°). The EADI will annunciate HDG 1/2 in green along the top. Automatic reduced 1/2 bank angle change will occur at 34,275 feet MSL during climbs. During descents, the bank angle will change to Full bank values passing through 34,275 feet MSL.

Standby (STBY)

Utilizing this mode will cause all Flight Director modes to cancel. The autopilot, if engaged, will remain engaged but revert to basic pitch and roll hold.

Flight Level Change (FLC)

Commands the system to maintain the current indicated airspeed or mach and altitude hold cancels if engaged. FLC displayed in green along the top of the EADI and speed command displayed on the lower left face of the EADI. Speed commands may be changed by using the autopilot pitch wheel (Figure 16-2) or the control wheel TCS button (Figure 16-5) and manually maneuvering the aircraft. Maneuvering the aircraft toward an altitude set in the AL-801 Altitude Preselect Controller will cause the ALT SEL annunciator to display in white along the top of the EADI. ALT SEL will display in green and the FLC light will drop out during the transition to capture the altitude set in the AL-801 controller.



Figure 16-5. AP/TRIM/NWS DISC and TCS Buttons

Change Over (C/O)

Allows the speed commands to be changed from IAS to MACH and vice versa. IAS commands automatically change to MACH in a climb above 34,275 feet or when 0.67 M is achieved. During descents, the reference changes to IAS from MACH when 300 knots is achieved. The speed commands may be changed back to the previous reference (IAS or MACH) by depressing the C/O button.



Vertical Speed (VS)

Selects the system to maintain the current vertical speed and to change to a new vertical speed command by utilizing the AP pitch wheel or the TCS button. VS is annunciated in green along the top of the EADI and the VS command is displayed in the lower left corner of the EADI.

Touch Control Steering (TCS)

The touch control steering (TCS) switches on the cockpit control wheels enables the aircraft to be maneuvered manually during autopilot operation without cancellation of any selected Flight Director modes. Depressing the TCS button (Figure 16-5) causes autopilot interruption only while it is depressed; releasing the button re-engages the autopilot. The TCS button may be used to establish a new reference for the Flight Director operating in the indicated airspeed (IAS) mode, mach airspeed (M) mode, vertical speed (VS) mode, or altitude hold (ALT) mode. If the flight Director is not engaged in any vertical modes, then the TCS button may be used as a pitch sync reference for the command bar.

Emergency Descent Mode

This mode is automatically initiated by an input from a cabin pressure altitude sensor. The parameters for this mode to actuate are: aircraft altitude of 34,275 feet or higher, cabin altitude of 13,500 feet and the autopilot engaged. This mode disengages any previously coupled Flight Director modes. The red EMER DESCENT annunciator panel light illuminates and the MASTER WARNING RESET switch-lights flash.

The autopilot will initiate a pitch down to capture and maintain $V_{MO} - 10$ KIAS and roll left to a 35 degree bank angle for approximately 48 seconds to clear enroute airways.

The autopilot must be disengaged to cancel the emergency descent mode before reselecting normal autopilot/flight director operation. If the autopilot remains engaged during the emergency descent mode, the autopilot will command a flare to level at 15,000 feet.

NOTE

This altitude cannot be changed on the altitude preselect controller unless the autopilot has been disengaged (which will cancel the Emer Descent Mode).

ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

The standard flight instrument configuration consists of dual tube Honeywell ED-800 electronic displays on the pilot and copilot instrument panels. These 5-inch by 6-inch cathode ray tubes are identical and interchangeable (Figure 16-3). The top tubes are used for electronic attitude director indicators (EADIs) and have conventional inclinometers attached to the lower bezel of the display. The lower tubes are used as electronic horizontal situation indicators (EHSIs).

Associated with each display system (pilot and copilot) is a Symbol Generator and a Display Controller (DC-810) (Figure 16-6). The heart of the EFIS system are the symbol generators, which receive and process all air plane sensor inputs. The data is then transmitted to the EFIS display tubes. The pilot and copilot control display formatting with the display controllers.

A common Instrument Remote Controller located on the center pedestal directly above the pilot display controller interfaces with the symbol generators to provide remote heading and course selection.

The EFIS system provides for display integration, flexibility and redundancy. The flight crew has essential display information, navigation, air data commands, attitude, and caution/failure warning systems integrated into their prime viewing areas. Each symbol generator is capable of driving four ED-800 Electronic Displays. If one side symbol generator fails, the remaining symbol generator will drive displays on both sides.

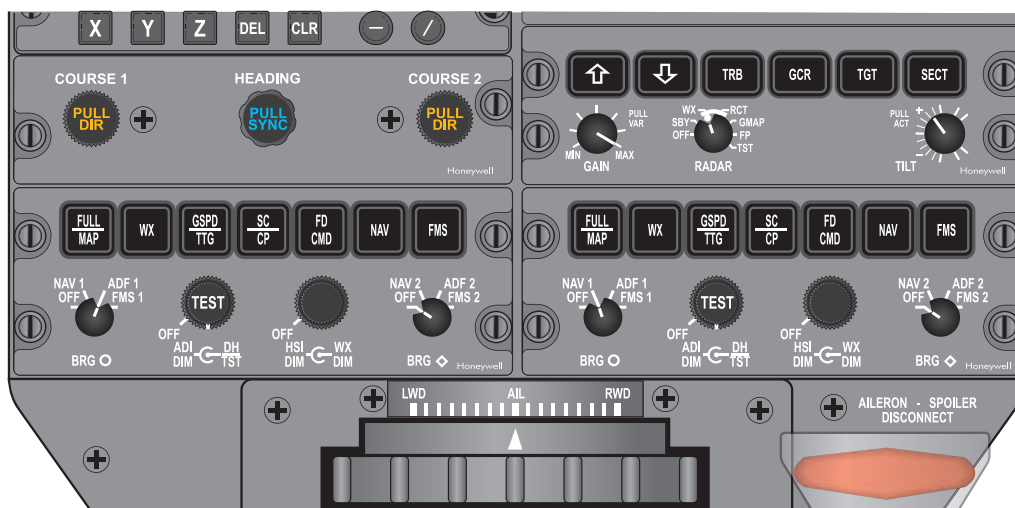
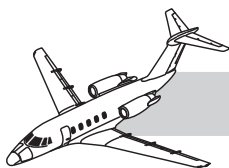


Figure 16-6. Display Controllers

Composite Mode

If one ED-800 display tube fails, a composite attitude/heading display format can be displayed on the remaining display tube on that side (Figure 16-7). This is accomplished by turning the ADI or HSI dim control OFF on the display controller of the failed display tube (Figure 16-6).

EFIS Reversionary

EFIS reversionary select buttons are located on the left side of the pilot instrument panel (Figure 16-8) and on the copilot lower switch panel (Figure 16-9). The reversionary select buttons allow the pilots to select alternate sources of symbol generators, heading, and attitude sources. When alternate sources are selected, one source driving both sides, the source annunciator on the display tubes will be amber in color, i.e., MAG 1 or 2, ATT 1 or 2, and/or SG 1 or 2.

NOTE

Consult the Honeywell SPZ-8000 DIFCS Pilot Manuals for the Citation VII and section III, Instrumentation and Avionics, of the *Airplane Operating Manuals* for detailed operating instructions of the Flight Guidance system.

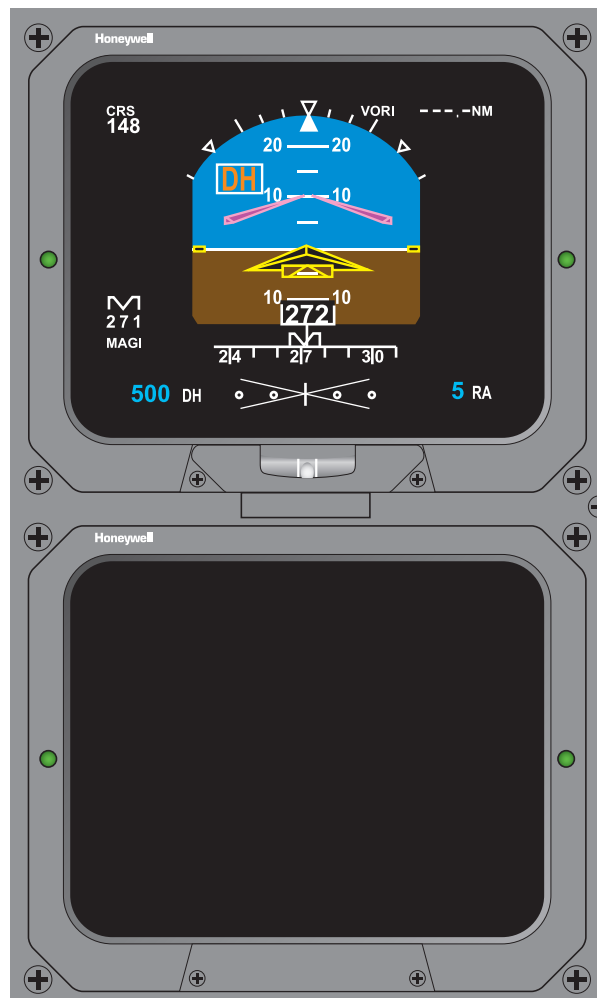


Figure 16-7. EFIS Tubes Composite

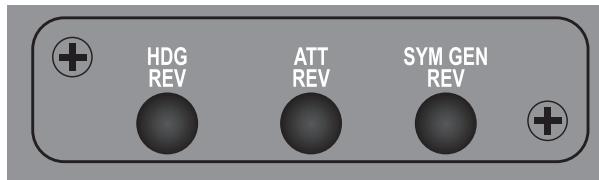


Figure 16-8. Pilot Reversionary Switch Panel



Figure 16-9. Copilot Lower Instrument Switch Panel

Multifunction Display System (MFD) Optional

An optional MDZ-816 Multifunction Display System (MFD) is installed on the lower center instrument panel (Figure 16-10). The MFD display serves as a radar indicator or as a backup to the EFIS system. The MFD symbol generator can be used to backup the EFIS symbol generators. The MFD display tube can also be used to back up the EHSI display tubes. The MFD system expands on the navigation mapping capability of the EFIS system.



Figure 16-10. MFD EFIS Tube

The MFD system is controlled by an MFD controller installed on the center pedestal (Figure 16-11). The controller is used to select various modes of operation: MAP, PLAN, weather, checklist (normal and emergency), and EFIS backup modes. Additional joy sticks are installed on both the pilot and copilot forward side console armrests to operate the checklist mode.

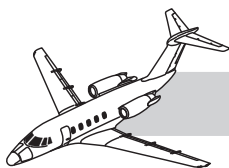


Figure 16-11. MFD Controller

ATTITUDE AND HEADING REFERENCE SYSTEM (AHRS)

The dual AHRS is the primary source of attitude and heading information. Each AHZ-600 Attitude and Heading Reference System (AHRS) consists of the following components:

- AH600 strapdown Attitude and Heading References Unit (AHRU)
- CS-412 Dual Remote Compensator



- FX-600 Thin Flux Valve
- External control switches

The characteristics of a strapdown gyro system are:

- Not gimbal mounted
- Spinning mass follows the airframe axis
- Output signals are rate sensitive

The dual AHZ-600s are mounted in the nose avionics bay. It is an inertial sensor system which provides aircraft attitude, heading, and flight dynamics information to the EFIS and MFD displays, flight control, weather radar, and various other aircraft systems and instruments. The rate gyros are strapped down to the principal aircraft axis. A digital computer mathematically integrates the rate data to obtain heading, pitch, and roll information. The Thin Flux Valve (FX-600), three accelerometers (one for each principle aircraft axis), and TAS from the Digital Air Data Computer (DADC), provide long-term references.

Mode of Operation

The standard operating modes are “Normal” for attitude and “Slaved” for heading. The accelerometers and the digital computer compensate for acceleration errors in the normal mode for the attitude system. The heading mode in slave is provided a magnetic compass reference from the flux valves. The AHRS system will revert to “basic” mode if the digital air data computer (DADC) true airspeed output becomes invalid. The respective AHRS BASIC 1 or 2 annunciator, located on the pilots lower instrument panel next to the EHSI, will illuminate. The “BASIC” attitude mode is similar in operation to a conventional attitude gyro system which is subject to drift and acceleration errors. With the AHRS in the slaved heading mode, a MAG 1 or 2 will be annunciated on the EHSI.

A dot “o” and a cross “+” heading sync indicator also is displayed on the top of the EHSI. A loss of magnetic reference will cause a HDG flag to display on the respective EHSI and the opposite RMI. ATT Reversionary, as previ-

ously discussed, may be selected to allow the remaining AHRS to provide magnetic heading, or the GYRO SLAVE MAN–AUTO button (Figures 16-9 and 16-12) can be selected. Selecting MAN will remove the HDG flag and the heading sync indicator from the EHSI. The system is now in a gyro stabilized mode and requires manual updating from a known correct heading source such as the magnetic compass. This can be accomplished by means of the GYRO SLAVE LH–RH switch (Figures 16-9 and 16-12).



Figure 16-12. Pilot Lower Instrument Switch Panel

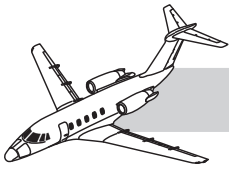
AHRS Auxiliary Power

The AHRS has an auxiliary battery pack mounted in the nose avionics bay, which provides power to the system during power transients and short power interruptions. If normal system DC power to the AHRS is lost, the AHRS will automatically switch to the auxiliary battery pack and the appropriate amber AHRS AUX PWR annunciator on the upper instrument panel will illuminate.

NOTE

The auxiliary battery pack must be satisfactorily tested prior to takeoff.

The AHRS and the standby gyro have separate battery packs. Both are energized by the standby gyro switch (Figure 16-12).



AHRS Preflight Test

An automatic self test occurs for five seconds upon initial power application. The following EFIS displays are present during the test:

EADI

- ATT Flag displays for 2.5 seconds
- 10 degrees pitch up
- 20 degrees right bank

EHSI

- HDG Flag displays for 2.5 seconds
- A North heading, turning East at 3 degrees per second
- AHRS BASIC 1/2 annunciator ON

NOTE

When EFIS and AHRS are powered up simultaneously, the test will not be visible due to warm up time of the EFIS displays.

AHRS Initialization

The AHRS requires approximately three minutes upon initial power application for alignment. The aircraft must not be moved until alignment is complete. Normal preflight activities, aircraft loading, and wind gusts will not affect the initialization process. Until alignment is complete, ATT and HDG flags will be displayed on the EFIS tubes, and the COMPTR WARN annunciator on top of the instrument panels will be illuminated. Time remaining for alignment may be checked by selecting the appropriate VERT GYRO FAST NORMAL selector switch to FAST. This will cause the EHSI compass to slew between 180° and 001° and become a timer (each degree mark represents one second). For example, 120° would represent two minutes until alignment complete. When the AHRS times out the HDG and ATT flags will disappear, the COMPTR WARN annunciator will extinguish and the EHSI compass will align to the aircraft magnetic heading.

Radio Magnetic Indicators (RMIs)

Dual RMI-36 radio magnetic indicators are mounted on both instrument panels (Figure 16-13). ADF and VOR magnetic bearing information is displayed on each RMI. The single-bar bearing pointers display VOR 1 or ADF 1. The double-bar bearing pointers display VOR 2 or ADF 2. Push-type selector switches for each pointer are mounted on the lower case of the RMI. The compass card for each RMI is driven by the opposite-side AHRS.



Figure 16-13. Radio Magnetic Indicator

Emergency Power Conditions

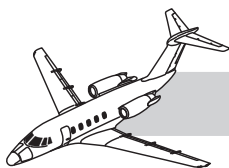
Operating on emergency DC electrical power only, the pilot RMI is operational (driven by the No.2 AHRS). During a loss of both inverters (normal DC power still available) the pilot RMI compass card is frozen, but the bearing pointers will point to ADF stations only (relative bearing only).

AIR DATA SYSTEM ADZ-810

General

The ADZ-810 Air Data System includes the following equipment:

- Dual ADZ-810 Digital Air Data Computers (DADC)



- Dual SI-225 Mach/Airspeed Indicator
- AL-801 Altitude Preselect Controller
- DS-125 A TAS/SAT/TAT Indicator

The AZ-810 DADCs receive both digital and analog inputs, process this data, and supply both digital and analog outputs. The DADC receives pitot-static pressures and total air temperature inputs for computing standard air data functions. The DADCs provide data for the altimeters, mach/airspeed indicators, transponders, EFIS/MFD symbol generators, flight guidance computers, optional flight data recorder, and other components of the flight control system. The AL-801 Altitude Preselect Controller provides the flight crew a system for selecting and displaying altitude reference for altitude alerting and altitude preselect functions. The DS-125A TAS/SAT/TAT indicator displays the true airspeed, total air temperature, and selectable static air temperature.

SI-225 Mach/Airspeed Indicators

Each SI-225 Mach/Airspeed indicator (Figure 16-14) is a servoed instrument featuring a mach counter display and an IAS pointer display. The maximum allowable V_{MO} and M_{MO} speeds are computed by the DADCs. A maximum allowable airspeed (barber pole) pointer will move to indicate maximum airspeeds with respect to altitude. Both indicators are fully electric (26 VAC) and display OFF flags during periods of power failure.

Both indicators are equipped with a selector switchlight (VMO SCH SELECT/ALTN) to select the standard zero fuel weight of 16,500 pound maximum airspeed limits or the alternate zero fuel weight of 15,350 pounds or less maximum airspeed limits. The selection VMO SCH SELECT (GREEN) or ALTN (AMBER) determines the speeds when the Mach/airspeed warning will sound.

Four movable indices (bugs) of different colors are located on the bezel glass ring for placement to critical airspeeds.



Figure 16-14. Airspeed Indicator and VMO Select Switch

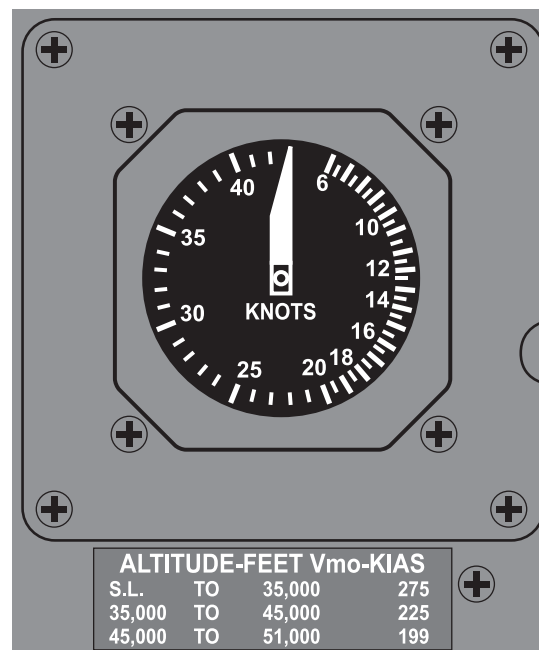


Figure 16-15. Standby Airspeed Indicator

Altimeters

Both altimeters are driven from their respective digital air data computer. They are electrically (26 VAC) servoed counter/pointer displays of



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FlightSafety[®]
international

barometrically corrected altitude. The pilot altimeter is both DADC driven or may be selected as a completely pneumatic altimeter from its own separate static ports (Figure 16-16). The copilot altimeter is completely electrically (26 VAC) driven (Figure 16-17).



Figure 16-16. Pilot Altimeter



Figure 16-17. Copilot Altimeter

If the pilot 26 VAC power is lost, the pilot altimeter will fail to the pneumatic mode of operation. If the altimeter fails to the pneumatic or is selected to pneumatic by the CADC/PNEU knob on the instrument, a PNEU flag (yellow) will appear on the face. In pneumatic

mode, automatic correction for instrument calibration does not occur.

Both altimeters have altitude encoding capability and communicate altitude data to the transponders. A switch labeled XPDR ALT PRI-SEC is normally located on the right side of the center instrument panel to control selection of the pilot or copilot altimeter for altitude encoding.

True Airspeed/ Temperature Indicator

The DS-125 TAS/TAT/SAT Indicator provides a digital display of both TAS (true airspeed) and TAT (total air temperature) or SAT (static air temperature) as generated by the DADC (digital air data computer) (Figure 16-18). The indicator normally displays true airspeed and total air temperature. Pressing the SAT button changes the temperature display to static air temperature as the button is held.

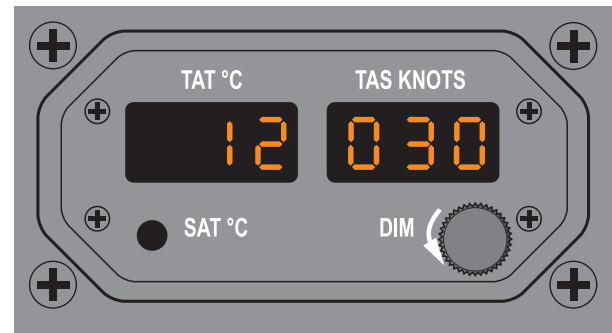


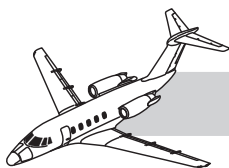
Figure 16-18. TAS/TAT/SAT Indicator

Altitude Preselect Controller

The AL-801 Altitude Preselect Controller provides a means for setting the desired altitude reference for altitude alerting and altitude preselect.

Altitude Preselect

The desired altitude is selected by turning the "set" knob. Altitude preselect is initiated by maneuvering the aircraft toward the preselected altitude. ALT SEL will be annunciated in white on the top of the EADI. Either FLC or VS mode can be engaged for the flight director command bar reference.



Altitude Alert

As the aircraft reaches a point 1,000 feet from the preselected altitude, a warning horn sounds for 1 second and a warning light on the altimeters illuminates. The light remains illuminated until the aircraft is 250 feet from the selected altitude. If the aircraft deviates more than 250 feet from the selected altitude, the warning light reilluminates and the 1 second warning will sound.

If the flight director is engaged in FLC airspeed or mach speed, or VS while the aircraft is climbing or descending, the flight director will command a flare to transition on the preselected altitude. During this transition, FLC or VS will drop out on the EADI and the ALT SEL annunciator will change from white to green. When the aircraft is established on the selected altitude, ALT SEL will disappear and ALT will annunciate in green on the EADI and the flight director will now command “altitude hold.”

Vertical Speed Indicators (VSI)

The pilot and copilot instantaneous vertical speed indicators are displayed on the right side of the electronic attitude director indicators (EADIs). A green FPM below the scale identifies the vertical speed scales. Each respective DADC drives the respective VSIs. A digital numerical vertical speed is displayed in feet-per-minute (FPM) in a box in the center of the scale. An analog depiction of vertical speed is displayed by a green line stretching vertically in the direction of the climb or descent.

The indicators indicate vertical velocity from 0 to 6,000 feet per minute in each direction. There is zero time lag between aircraft displacement and instrument indication. The vertical speed tape and the digital readout have a dead band of 75 FPM to reduce activity of the digital readout.

Optional Analog Vertical Speed Indicator

An optional VSI may be installed as a backup for the standard VSI on the EADIs. The analog

(round) VSI is normally on the pilot instrument panel and is electrically powered (26 VAC) from the number 1 (pilot) inverter.

STANDBY FLIGHT INSTRUMENTS

Standby Attitude Gyro

The standby attitude indicator system consists of a gyro horizon and an emergency power supply. The gyro horizon, mounted high on the pilot instrument panel to the left of the annunciator panel, is powered directly from the emergency power supply. Power to the gyro is controlled by the STBY GYRO switch, with ON, OFF, and TEST positions, located on the pilot lower instrument panel (Figure 16-12). An emergency battery pack in the nose avionics compartment is the emergency source of power for the standby gyro if main DC bus power is lost (generators OFF and battery switch EMER). In this case, the standby gyro will be operational on its self-contained battery. The battery pack also provides power for emergency instrument lighting for the pilot primary flight instruments.

In normal operation, the battery pack is continuously charged by the main DC electrical system and should be fully charged in the event of an electrical power failure. The STBY GYRO switch must be in the ON position for automatic transfer to emergency battery pack power. The amber light adjacent to the switch illuminates when the gyro is operating on its own power source (battery pack). A fully charged battery pack will provide 30 minutes of operating time. When the switch is held to the TEST position, a self-test of the emergency battery pack and associated electrical circuits is accomplished. The amber light adjacent to the switch illuminates if the test is satisfactory and the battery pack is fully charged. The green light illuminates with the switch ON and aircraft power operating the standby gyro and charging the emergency battery pack.

The standby gyro is caged by pulling the PULL TO CAGE knob and rotating it clockwise.

**CAUTION**

When uncaging, do not release the PULL TO CAGE knob suddenly so that it snaps back; this can damage the gyro.

Standby Airspeed Indicator

A standby pneumatic airspeed indicator is installed on the pilot instrument panel (Figure 16-15). The indicator requires no electrical power and receives inputs from the copilot pitot-static system.

NOTE

When operating with the standby airspeed indicator as primary, a new maximum V_{MO} schedule based on altitude must be adhered to. This schedule is posted on a placard directly below the indicator (Figure 16-15).

INSTRUMENT PANEL ANNUNCIATORS

Annunciator warning lights associated with the Integrated Flight Control System (DIFCS) are located directly above the pilot and copilot EADIs on the instrument panels. They are as follows:

FMS1 SX and FMS2 SX—Parallel track. Illuminates when FMS-1 or FMS-2 has been programmed for course guidance with respect to a course offset from, but parallel to, the leg shown on the CDU.

VLF1 Battery and VLF2 Battery—VLF/OMEGA Battery. Illuminates when the GNS-X RPU is being electrically powered by its own internal standby battery. Illuminates during periods of ships' power losses.

CMPTTR WARN—Each symbol generator for the pilot and copilot EFIS monitors its various raw data inputs. When the input data for the same type data exceeds certain parameters the annunciators illuminate. A symbol for the parameter exceeded is displayed on the EADI as follows:

PTT—Pitch attitude exceeds ± 6 degrees.

ROL—Roll attitude exceeds ± 6 degrees.

HDG—Heading data exceeds ± 6 degrees.

LOC—Localizer signals exceed 1/2 dot deviation.

GS—Glideslope signals exceed 2/3 dot deviation.

ATT—Pitch and roll attitude exceeds ± 6 degrees.

ILS—Localizer ± 40 mV glideslope ± 50 mV.

AP OFF—Illuminates when either the pilot or the FGS disengages the autopilot. A one second tone also sounds.

YD OFF—Illuminates when either the pilot or the FGS disengages the autopilot or the YD. The autopilot may be disengaged separately without disengaging the yaw damper by depressing the primary trim switch.

EFIS FAN—EFIS cooling fan. Illuminates if the pilot or copilot EFIS cooling fans or nose avionics bay cooling fans fail.

AHRS AUX PWR—AHRS auxiliary battery pack. AHRS 1 or 2 has transferred to the auxiliary battery AUX PWR due to or loss of DC power. Auxiliary battery power is limited to 5 minutes.

Annunciator warning lights located on the center instrument panel are:

FAIL A RESET—Left Flight Guidance System (FGS) computer invalid. May be reset.

FAIL B RESET—Right Flight Guidance System (FGS) computer invalid. May be reset.

AIL TRIM L-R—FGS senses a steady-state load on the roll servo. Trim ailerons in the direction indicated.

ELEV TRIM UP-DN—FGS senses a steady-state load on the pitch servo. If this light does not clear automatically within a few seconds, autopilot should be disengaged.



Annunciator lights located on the pilot lower instrument panel are:

AHRS BASIC 1–2—True airspeed from one or both DADCs becomes invalid to the No. 1 and/or No. 2 AHRS. AHRS operation in the BASIC mode results in the attitude system (VG) subject to conventional drift and acceleration errors.

AFCS A ON—Left Flight Guidance computer (A-channel) is being used as the controlling system.

AFCS B ON—Right Flight Guidance computer (B-channel) is being used as the controlling system. The switchlights may be depressed to enable the pilot to select the controlling system, left or right.

MFD FAN—MFD cooling fan. Illuminates if the fan fails.

FLIGHT GUIDANCE SYSTEM: SNs 0179–0199 AND 0203–0206

GENERAL

The standard flight instrumentation consists of the SPZ-8000 digital integrated flight control system. The SPZ-8000 consists of two independent electronic flight instrument systems, one independently controlled system on each (pilot and copilot) instrument panel (Figure 16-19). Each system (pilot and copilot) may be used in conjunction with the single flight guidance controller and the single remote instrument controller to fly the aircraft. The crew may use the system to fly the aircraft manually using the flight director, or automatically using the flight guidance controller. The system meets Category II equipment requirements when flown with the autopilot engaged.



Figure 16-19. SPZ-8000 DIFCS Flight Deck: SNs 0179–0199 and 0203–0206



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The SPZ-8000 includes the DFZ-800 Flight Guidance System (FGS), EDZ 816 Electronic Flight Instrument System (EFIS), AHZ-600 Attitude and Heading Reference System (AHRs), and ADZ-810 Air Data System.

The SPZ-8000 is a complete automatic control system. Flight path commands are generated by the FZ-800 flight guidance computers, which integrate attitude and heading from the AHRs, air data from the ADZ-810 system, navigation from various navigation components, and the EFIS all via an avionics standard communications bus (ASCB).

The system requires primarily 28 VDC power for operation. The air data computers, altimeters, and airspeed display require 26 VAC. The weather radar system also requires 26 VAC for antenna stabilization.

DFZ-800 FLIGHT GUIDANCE SYSTEM

The DFZ-800 flight guidance system consists of an integrated autopilot/flight director, dual A-channel/B-channel flight guidance computers, and three flight control servos (elevator, aileron, and rudder). Dual AHRs and air data computers provide fail-safe redundancy along with various sensor fail-safe characteristics. A single GC-810 flight guidance controller is used to engage the autopilot and select operating modes for the flight director (Figure 16-20).

Power-Up Test

Upon initial application of DC power, the flight guidance system initiates a self-test for approximately 30 seconds. If the test is satisfactory, the AFCS A-switchlight, located on the forward throttle quadrant, illuminates (Figure 16-20). The A-channel lights adjacent to the left side of the AP and YD engage switches also illuminate (Figure 16-20). If a failure is detected in the A-channel flight guidance computer, both the AFCS B-switchlight and the B-channel lights on the controller illuminate.

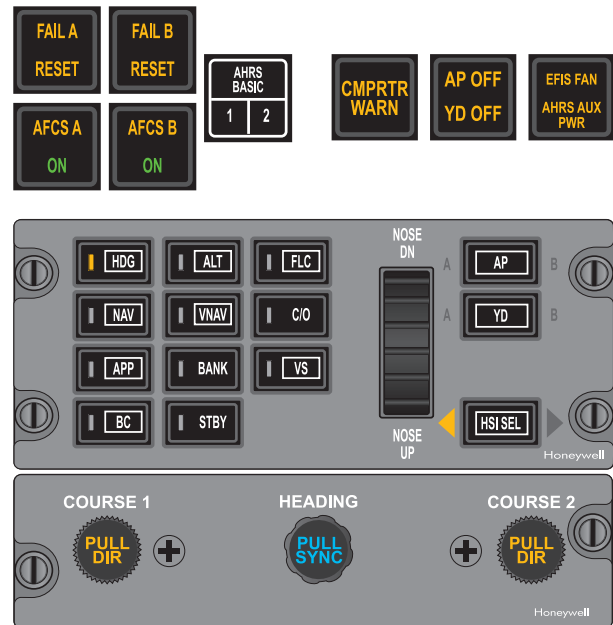
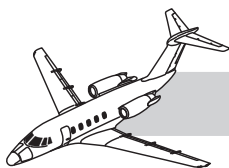


Figure 16-20. Flight Guidance and Instrument Remote Controllers and Associated Annunciators

If a failure is detected in either flight guidance computer, the appropriate FAIL A or B RESET annunciator, on the center instrument panel, illuminates. The failed computer may be reset by pushing and releasing the appropriate FAIL A or B RESET switchlight. If the reset is satisfactory, the appropriate FAIL A or B annunciator extinguishes in approximately two seconds.

If a failure occurs within a flight guidance computer that disables the flight director functions, the FD FAIL annunciator on the EADI illuminates. This malfunction also causes the flight director command bar on the EADI to disappear.

If both flight guidance computers fail, the second computer failure disengages the DFZ-800 flight guidance system and the autopilot. The AP OFF annunciators above the EADIs illuminate (Figure 16-20), and the autopilot warning horn sounds.



Autopilot Functions

The autopilot switches on the controller provide for engagement and disengagement of the autopilot and yaw damper functions. They also provide manual control of the autopilot through the PITCH wheel and the HEADING knob on the instrument controller (Figure 16-20).

The push-on AP and YD switches are illuminated when engaged. The autopilot and yaw damper may be disengaged by pushing the AP switch. When engaged, the AP and YD switches may be disengaged by various methods. To simultaneously disconnect the AP and YD, push the red AP/TRIM/NWS DISC button on the control wheels. The autopilot can also be disengaged as follows:

- Pressing the go-around button on the throttles (provided flight director is engaged)
- Selecting the AOA/flap test with the rotary test switch
- Activating primary trim (does not disengage the yaw damper)
- Activating the AHRS manual-slave LH–RH switch
- Activating the AHRS fast-erect switch

NOTE

The autopilot cannot be engaged on the ground.

HSI Select

The HSI SEL button (Figure 16-20) is used to select either the pilot or copilot EHSI and DADC data for lateral and vertical guidance to both flight guidance computers. The arrow lights indicators adjacent to the selector switch displays which side is controlling the flight guidance computers. During an ILS approach, captured inbound, with both navigation receivers turned to the same ILS frequency and the aircraft within 1,200 feet of the terrain, both arrows illuminate. If one ILS receiver fails, the arrow on that side extinguishes, the other arrow remains illuminated, and the sys-

tem automatically selects the remaining operating side for display.

When the HSI select button is used to transfer to the cross-side EHSI and DADC, all flight director modes are canceled. The flight director must be reprogrammed for the new side for operation.

NOTE

The arm and/or capture status of the flight director is annunciated only on the EADI.

Flight Director—Modes and Annunciations

Heading (HDG)

HDG is annunciated in green along the top of the EADI (Figure 16-21) and commands the flight guidance computer to track inputs of the heading bug on the EHSI (Figure 16-21). The heading bug is controlled by the HEADING knob on the instrument controller (Figure 16-20). While in this mode, a lower angle-of-bank limit (17°) may be selected with the BANK button (Figure 16-20).

Navigation (NAV)

VOR, LOC, AZ, and LNAV are annunciated along the top of the EADI (white—arm, green—capture). The flight guidance computer is programmed to arm, capture, and track the selected navigation signals from the navigation receivers.

Approach (APP)

APP increases the signal gain to arm and capture lateral deviation signals for VOR, LOC, and AZ and both lateral and deviation signals for ILS and MLS. The EADI will annunciate VAPP with a VOR frequency tuned

Back Course (BC)

BC commands the flight guidance computer to track the localizer back course and reverses the directional input for the command bar.

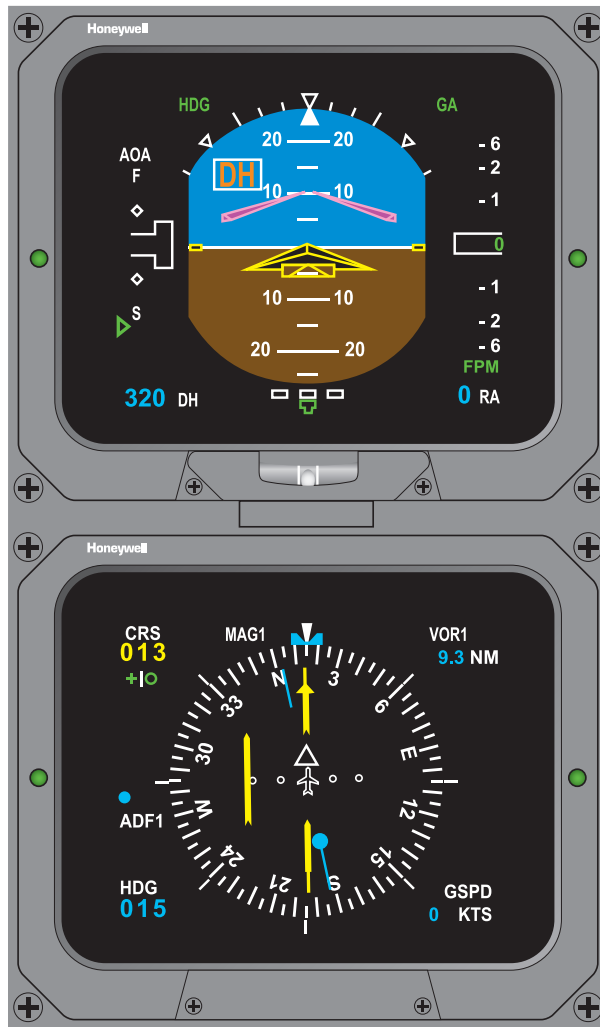


Figure 16-21. Electronic Flight Instrument System

NOTE

Ensure that the inbound FRONT course is set in the course window on the EHSI (Figure 16-21). It is set by the COURSE 1/2 knobs on the instrument controller (Figure 16-20).

BC is displayed in white or green, as appropriate, along the top of the EADI.

Altitude (ALT)

ALT commands the system to hold the current aircraft altitude at the time of engagement.

ALT is annunciated in green along the top of the EADI. Capturing the altitude displayed on the AL-801 Altitude Preselect Controller allows the system to maintain the displayed altitude (Figure 16-22).

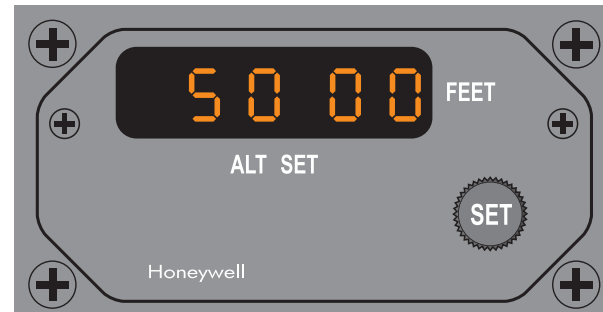


Figure 16-22. AL-801 Altitude Preselect Controller

Vertical Navigation (VNAV)

VNAV provides vertical path guidance from a compatible long-range navigation system (Honeywell FMS only).

Bank (BANK)

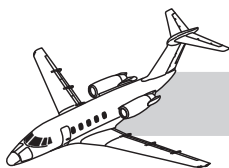
Engaging this mode commands the flight guidance computer to reduce the maximum bank angle when in the HDG mode (17°). The EADI annunciates HDG 1/2 in green along the top. Automatically reduced 1/2 bank angle change occurs at 34,275 feet MSL during climbs. During descents, the bank angle changes to full bank values passing through 34,275 feet MSL.

Standby (STBY)

Utilizing this mode causes all flight director modes to cancel. The autopilot, if engaged, remains engaged but reverts to basic pitch and roll hold.

Flight Level Change (FLC)

Selecting FLC commands the system to maintain the current indicated airspeed or Mach, and altitude hold cancels if engaged. FLC is displayed in green along the top of the EADI, and the speed command is displayed on the lower left face of the EADI. Speed commands



may be changed by using the autopilot pitch wheel (Figure 16-20) or the TCS buttons on the control wheels (Figure 16-23) and manually maneuvering the aircraft. Maneuvering the aircraft toward an altitude set in the AL-801 Altitude Preselect Controller (Figure 16-22) causes the ALT SEL annunciator to display in white along the top of the EADI. ALT SEL displays in green and the FLC light drops out during the transition to capture the altitude set in the AL-801 controller.



Figure 16-23. AP/TRIM/NWS DISC and TCS Buttons

Change Over (C/O)

Selecting C/O allows the speed commands to be changed from IAS to Mach and vice versa. IAS commands automatically change to Mach in a climb above 34,275 feet or when 0.67 M is achieved. During descents, the reference changes to IAS from Mach when 300 knots is achieved. The speed commands may be changed back to the previous reference (IAS or Mach) by depressing the C/O button.

Vertical Speed (VS)

VS selects the system to maintain the current vertical speed and to change to a new vertical speed command by utilizing the AP pitch wheel or the TCS buttons. VS is annunciated in green along the top of the EADI, and the VS command is displayed in the lower left corner of the EADI.

Touch Control Steering (TCS)

The touch control steering (TCS) buttons on the cockpit control wheels enable the aircraft to be maneuvered manually during autopilot operation without cancellation of any selected flight director modes. Depressing the TCS button (Figure 16-23) causes autopilot interruption only while it is depressed; releasing the button re-engages the autopilot. The TCS button may be used to establish a new reference for the flight director operating in the indicated airspeed (IAS) mode, Mach airspeed (M) mode, vertical speed (VS) mode, or altitude hold (ALT) mode. If the flight director is not engaged in any vertical modes, then the TCS button can be used as a pitch sync reference for the command bar.

Emergency Descent Mode

This mode is automatically initiated by an input from a cabin pressure altitude sensor. The parameters for this mode to actuate are the following:

- Aircraft altitude of 34,275 feet or higher
- Cabin altitude of 13,500 feet
- The autopilot engaged

This mode disengages any previously coupled flight director modes. The red EMER DESCENT annunciator panel light illuminates, and the MASTER WARNING RESET switchlights flash.

The autopilot initiates a pitchdown to capture and maintain $V_{MO} - 10$ KIAS and roll left to a 35° bank angle for approximately 48 seconds to clear enroute airways.

The autopilot must be disengaged to cancel the emergency descent mode before reselecting normal autopilot/flight director operation. If the autopilot remains engaged during the emergency descent mode, the autopilot commands a flare to level at 15,000 feet.



ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

General

The standard flight instrument configuration consists of dual-tube Honeywell ED-800 electronic displays on the pilot and copilot instrument panels. These 5-inch by 6-inch cathode ray tubes are identical and interchangeable (Figure 16-21). The top tubes are used for electronic attitude director indicators (EADIs) and have conventional inclinometers attached to the lower bezel of the display. The lower tubes are used as electronic horizontal situation indicators (EHSIs).

Associated with each display system (pilot and copilot) is a symbol generator and a display controller (DC-810) (Figure 16-24). The heart of the EFIS system is the symbol generators, which receive and process all aircraft sensor inputs. The data is then transmitted to the EFIS display tubes. The pilots control display formatting with the display controllers.

A common instrument remote controller, located on the center pedestal directly above the pilot display controller (Figure 16-24), interfaces with the symbol generators to provide remote heading and course selection.

The EFIS system provides for display integration, flexibility, and redundancy. The flight crew has essential display information, navigation, air data commands, attitude, and caution/failure warning systems integrated into their prime viewing areas. Each symbol generator is capable of driving four ED-800 electronic displays. If one side symbol generator fails, the remaining symbol generator will drive the displays on both sides.

Composite Mode

If one ED-800 display tube fails, a composite attitude/heading display format can be displayed on the remaining display tube on that side (Figure 16-25). This is accomplished by turning the ADI or HSI dim control OFF on the display controller or the failed display tube (Figure 16-26).

Reversionary Mode

EFIS reversionary select buttons are located on the left side of the pilot instrument panel (Figure 16-27) and on the copilot reversionary switch panel (Figure 16-28). The reversionary select buttons allow the pilots to select alternate sources of symbol generators, heading, and attitude. When alternate sources are selected, one source driving both sides, the source

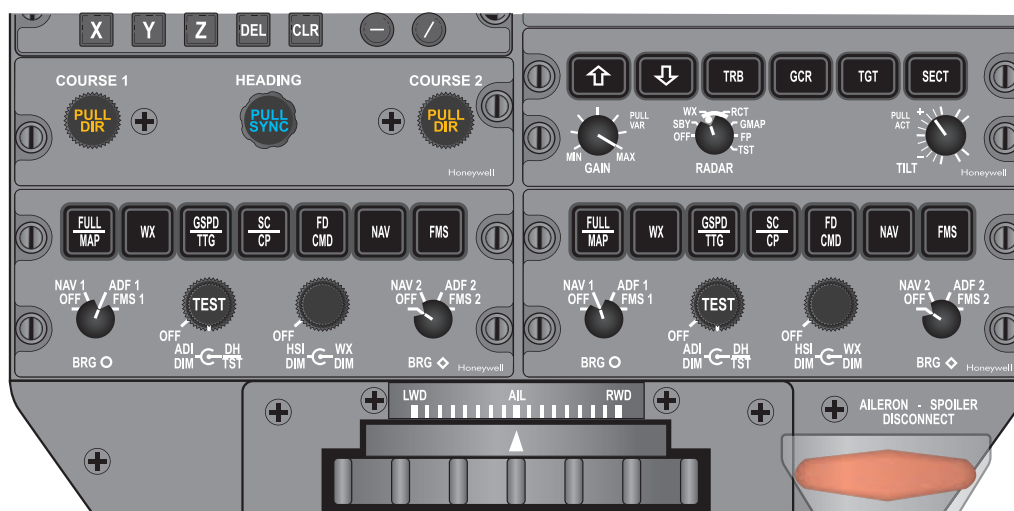


Figure 16-24. Display Controllers

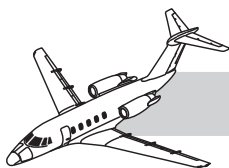


Figure 16-25. ED-800 Display Tube (Composite Format)

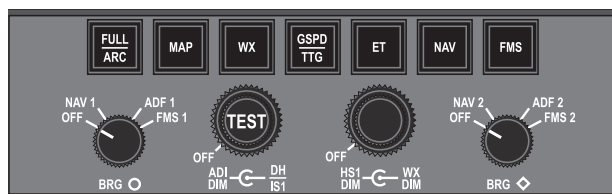


Figure 16-26. DC-810 Display Controller



Figure 16-27. Pilot and Copilot Reversionary Switch Panel



Figure 16-28. Copilot Lower Switch Panel

annunciator on the display tubes will be amber in color, i.e., MAG 1 or 2, ATT 1 or 2, and/or SG 1 or 2.

Multifunction Display System (MFD) (Optional)

An optional MDZ-816 multifunction display system (MFD) is installed on the lower center instrument panel (Figure 16-29). The MFD display serves as a radar indicator or as a backup to the EFIS system. The MFD symbol generator can be used to back up the EFIS symbol generators.

The MFD display tube can also be used to back up the EHSI display tubes. The MFD system expands on the navigation mapping capability of the EFIS system.

The MFD is controlled by an MFD controller installed on the center pedestal (Figure 16-30). The controller is used to select various modes of operation: MAP, PLAN, WEATHER, CHECKLIST (normal and emergency), and EFIS backup modes. Additional joysticks are installed on both the pilot and copilot forward side console armrests to operate the checklist mode.

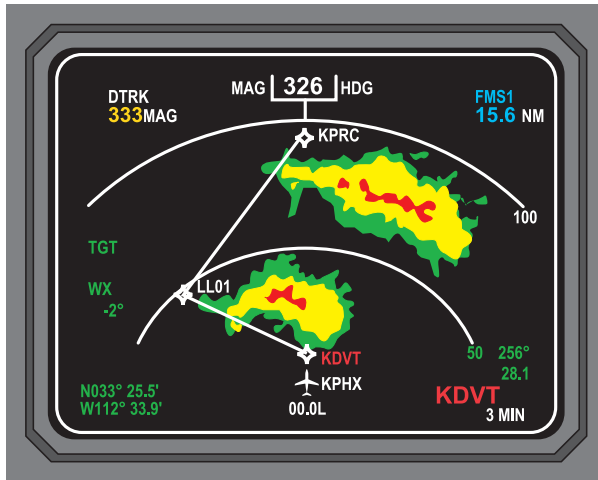


Figure 16-29. MFD Display Tube



Figure 16-30. MFD Controller

ATTITUDE AND HEADING REFERENCE SYSTEM (AHRS)

General

The dual AHRS is the primary source of attitude and heading information. Each AHZ-600 attitude and heading reference system (AHRS) consists of the following components:

- AH-600 strapdown attitude and heading reference unit (AHRU)
- CS-412 dual remote compensator
- FX-600 thin flux valve
- External control switches

The characteristics of a strapdown gyro system are the following:

- Not gimbal-mounted
- Spinning mass follows the airframe axis
- Output signals are rate-sensitive

The dual AHZ-600s are mounted in the nose avionics bay. It is an inertial sensor system which provides aircraft attitude, heading, and flight dynamics information to the EFIS and MFD displays, flight controls, weather radar, and various other aircraft systems and instruments. The rate gyros are strapped down to the principal aircraft axis. A digital computer mathematically integrates the rate data to obtain heading, pitch, and roll information. The thin flux valve (FX-600), three accelerometers (one for each principle aircraft axis), and TAS from the digital air data computer (DADC) provide long-term references.

Mode of Operation

The standard operating modes are “normal” for attitude and “slaved” for heading. The accelerometers and the digital computer compensate for acceleration errors in the normal mode for the attitude system. The heading mode in slave is provided a magnetic compass reference from the flux valves. The AHRS system reverts to “basic” mode if the digital air data computer (DADC) true airspeed output becomes invalid. The respective AHRS BASIC 1 or 2 annunciator, center pedestal forward of the throttles, illuminates. The BASIC attitude mode is similar in operation to a conventional attitude gyro system, which is subject to drift and acceleration errors.

When the AHRS is in the slaved heading mode, a MAG 1 or 2 is annunciated on the EHSI. A dot (o) and (+) heading sync indicator is also displayed on the top of the EHSI. A loss of magnetic reference causes a HDG flag to display on the respective EHSI and the opposite RMI. ATT reversionary, as previously discussed, may be selected to allow the remaining AHRU to provide magnetic heading, or the GYRO SLAVE MAN-AUTO button (Figures 16-28 and 16-31) can be selected. Selecting MAN removes the HDG flag and the heading sync indicator from the EHSI and replaces the MAG 1 or 2 symbol with a DG symbol. The system is now in a gyro-stabilized mode and requires manual updating from a known correct heading source such as the magnetic compass. This can be accomplished by means of the GYRO SLAVE LH-RH switch (Figures 16-28 and 16-31).

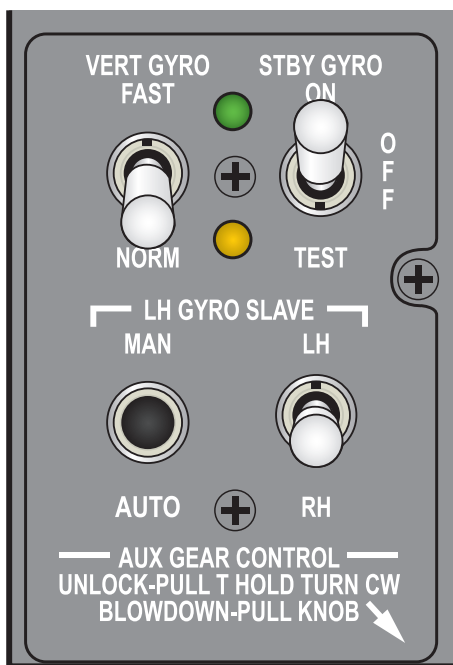
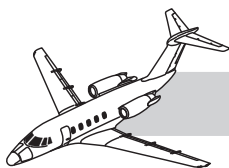


Figure 16-31. Pilot Lower Switch Panel

AHRS Auxiliary Power

The AHRS has an auxiliary battery pack mounted in the nose avionics bay, which provides power to the system during power transients and short power during power interruptions. If normal system DC power to the AHRS is lost, the AHRS automatically switches to the auxiliary battery pack, and the appropriate amber AHRS AUX PWR annunciator on the upper instruments panel illuminates.

NOTE

The auxiliary battery pack must be satisfactorily tested prior to takeoff.

The AHRS and the standby gyro have separate battery packs. Both are energized by the standby gyro switch (Figure 16-31).

AHRS Preflight Test

An automatic self-test occurs for five seconds upon initial power application. The following EFIS displays are present during the test.

EADI

- ATT flag displays for 2.5 seconds
- 10° pitch up
- 20° right bank

EHSI

- HDG flag displays for 2.5 seconds
- A north heading, turning east 3° per second
- AHRS BASIC 1/2 annunciator on

NOTE

When EFIS and AHRS are powered up simultaneously, the test will not be visible due to warm up time of the EFIS displays.

AHRS Initialization

The AHRS requires approximately three minutes upon initial power application for alignment. The aircraft must not be moved until alignment is complete. Normal preflight activities, aircraft loading, and wind gusts will not affect the initialization process. Until alignment is complete, ATT and HDG flags will be displayed on the EFIS tubes, and the COMPTR WARN annunciator on top of the instrument panels will be illuminated. Time remaining for alignment may be checked by selecting the appropriate VERT GYRO FAST-NORMAL selector switch to FAST (Figures 16-28 and 16-31). This causes the EHSI compass to slew between 180° and 001° and become a timer (each degree mark represents one second). For example, 120° would represent two minutes until alignment is complete. When the AHRS times out the HDG and ATT flags disappear, the COMPTR WARN annunciator extinguishes, and the EHSI compass aligns to the aircraft magnetic heading.

Radio Magnetic Indicators (RMIs)

Dual RMI-36 radio magnetic indicators are mounted on both instrument panels (Figure 16-32). ADF and VOR magnetic bearing infor-



mation is displayed on each RMI. The single-bar bearing pointers display VOR 1 or ADF 1. The double-bar bearing pointers display VOR 2 or ADF 2. Push-type selector switches for each pointer are mounted on the lower case of the RMI. The compass card for each RMI is driven by the opposite-side AHRS.



Figure 16-32. Pilot RMI (RMI-36)

Emergency Power Conditions

Operating on emergency DC electrical power only, the pilot RMI is operational (driven by the No. 2 AHRS). During a loss of both inverters (normal DC power still available) the pilot RMI compass card is frozen, but the bearing pointers will point to ADF stations only.

AIR DATA SYSTEM (ADZ-810)

General

The ADZ-810 Air Data System includes the following equipment:

- Dual AZ-810 Digital Air Data Computers (DADCs)
- Dual SI-225 Mach/Airspeed Indicators
- AL-801 Altitude Preselect Controller
- DS-125A TAS/TAT/SAT Indicator
- Altimeters

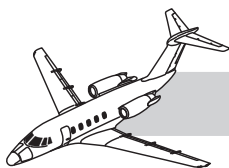
The AZ-810 DADCs receive both digital and analog inputs, process this data, and supply both digital and analog outputs. Each DADC receives pitot-static pressures and total air temperature inputs for computing standard air data functions. They provide data for the altimeters, Mach/airspeed indicators, transponders, EFIS/MFD symbol generators, flight guidance computers, optional flight data recorder, and other components of the flight control system. The AL-801 Altitude Preselect Controller provides the flight crew a system for selecting and displaying altitude reference for altitude alerting and altitude preselect functions. The DS-125A TAS/TAT/SAT indicator displays true airspeed, total air temperature, and selectable static air temperature.

SI-225 Mach/Airspeed Indicators

Each SI-225 Mach/Airspeed Indicator (Figure 16-33) is a servoed instrument featuring a Mach counter display and an IAS pointer display. The maximum allowable V_{MO} and M_{MO} speeds are computed by the DADCs. A maximum allowable airspeed (barber pole) pointer will move to indicate maximum airspeeds with respect to altitude. Both indicators are fully electric (26 VAC) and display OFF flags during periods of power failure.



Figure 16-33. Mach/Airspeed Indicator



Four movable indices (bugs) of different colors are located on the bezel glass ring for placement to critical airspeeds.

Altimeters

Both altimeters are driven from their respective digital air data computer. They are electrically (26 VAC) servoed counter/pointer displays of barometrically corrected altitude. The pilot altimeter (Figure 16-34) is DADC-driven or may be selected as a completely pneumatic altimeter from its own separate static ports. The copilot altimeter (Figure 16-35) is completely electrically (26 VAC) driven.



Figure 16-34. Pilot Altimeter



Figure 16-35. Copilot Altimeter

If the pilot 26 VAC power is lost, his altimeter fails to the pneumatic mode of operation. If the altimeter fails to pneumatic or is selected to pneumatic by the CADC/PNEU knob on the instrument panel, a PNEU flag (yellow) appears on the face. In pneumatic mode, automatic correction for instrument calibration does not occur.

Both altimeters have altitude encoding capability and communicate altitude data to the transponders. A switch labeled “XPDR ALT PRI-SEC” is normally located on the right side of the center instrument panel to control selection of the pilot or copilot altimeter for altitude encoding.

True Airspeed/ Temperature Indicator

The DS-125A TAS/TAT/SAT Indicator (Figure 16-36) provides a digital display of true airspeed (TAS) and total air temperature (TAT) or static air temperature (SAT) as generated by the DADC. The indicator normally displays true airspeed and total air temperature. Pressing the SAT button changes the temperature display to static air temperature as the button is held.

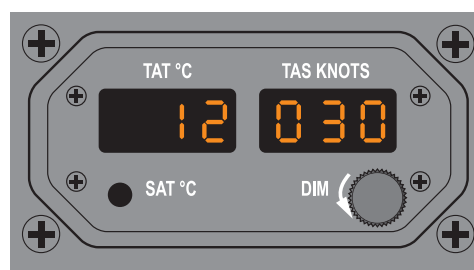


Figure 16-36. DS-125A TAS/TAT/SAT Indicator

Altitude Preselect Controller

The AL-801 Altitude Preselect Controller provides a means for setting the desired altitude reference for altitude alerting and altitude preselect (Figure 16-37).

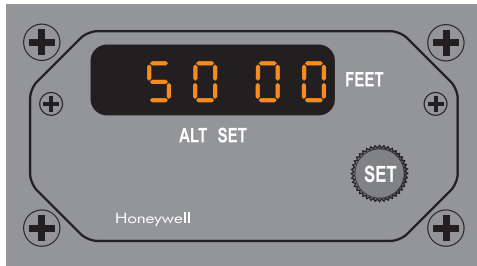


Figure 16-37. AL-801 Altitude Preselect Controller

Altitude Preselect

The desired altitude is selected by turning the SET knob. Altitude preselect is initiated by maneuvering the aircraft toward the preselected altitude. ALT SEL is annunciated in white on the top of the EADI. Either FLC or VS mode can be engaged for the flight director command bar reference.

Altitude Alert

As the aircraft reaches a point 1,000 feet from the preselected altitude, a warning horn sounds for one second, and a warning light on the altimeter illuminates. The light remains illuminated until the aircraft is 250 feet from the selected altitude. If the aircraft deviates more than 250 feet from the selected altitude, the warning light reilluminates and the one-second warning sounds.

If the flight director is engaged in FLC airspeed or Mach speed, or VS while the aircraft is climbing or descending, the flight director commands a flare to transition on the preselected altitude. During this transition, FLC or VS drops out on the EADI, and the ALT SEL annunciator changes from white to green. When the aircraft is established on the selected altitude, ALT SEL disappears, ALT annunciates in green on the EADI, and the flight director now commands “altitude hold.”

Analog Vertical Speed Indicators (VSIs)

Analog VSIs are installed on both the pilot and copilot instrument panels (Figure 16-38).

Each VSI is electrically powered (26 VAC) from its respective inverter, i.e., pilot VSI powered from the No. 1 inverter and the copilot VSI powered from the No. 2 inverter.

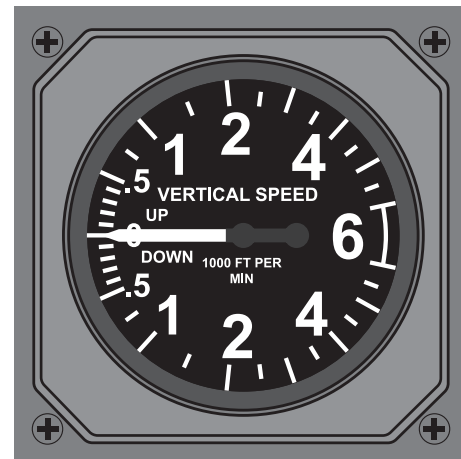


Figure 16-38. Analog VSI

STANDBY FLIGHT INSTRUMENTS

Standby Attitude Gyro

A standby attitude indicator is located on the pilot instrument panel (Figure 16-39). It normally operates on emergency DC electrical power through the STANDBY ATT IND circuit breaker on the right circuit breaker panel. Power to the gyro is controlled by the STBY GYRO switch, with ON, OFF, and TEST positions located on the pilot lower instrument panel (Figure 16-31). An emergency battery pack in the nose avionics compartment is an emergency source of power for the standby gyro if emergency DC bus voltage falls below minimum. This is indicated by an amber power-on light adjacent to the STBY GYRO switch, provided the switch is in the ON position. The battery pack also provides power for emergency instrument lighting for the pilot primary flight instruments.

The battery pack is continuously charged by the emergency DC electrical system and should be fully charged in the event of an electrical power failure. The STBY GYRO power switch must be in the ON position for

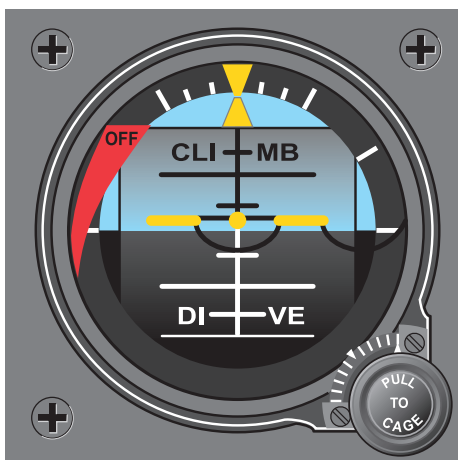
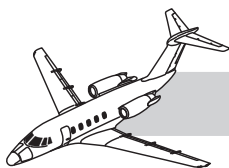


Figure 16-39. Standby Attitude Gyro Indicator

automatic transfer to emergency battery pack power. The amber light adjacent to the switch illuminates when the gyro is operating on its own power source (battery pack). When the switch is held to the TEST position, a self-test of the emergency battery pack and associated electrical circuits is accomplished. The amber light adjacent to the switch illuminates if the test is satisfactory and the battery pack is fully charged. The green light illuminates with the switch ON and aircraft power operating the standby gyro and charging the emergency battery pack.

The standby gyro is caged by pulling the PULL TO CAGE knob and rotating it clockwise (Figure 16-39).

CAUTION

When uncaging, do not release the PULL TO CAGE knob suddenly so that it snaps back; this can damage the gyro.

Standby Airspeed Indicator

A standby pneumatic airspeed indicator is installed on the pilot instrument panel (Figure 16-40). The indicator requires no electrical power and receives inputs from the copilot pitot-static system.

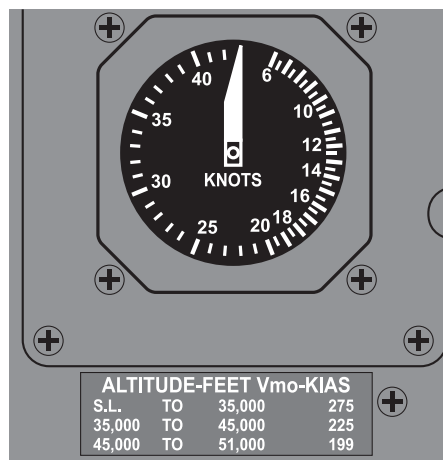


Figure 16-40. Standby Airspeed Indicator

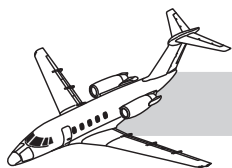
NOTE

When operating with the standby airspeed indicator as primary, a new maximum V_{MO} schedule based on altitude must be adhered to. This schedule is posted on a placard directly below the indicator (Figure 16-40).

SPZ-8000 EFIS SNs 0001–0178 (OPTIONAL)

General

Some aircraft in this serial number series are modified to incorporate an SPZ-8000 EFIS flight guidance system. These aircraft have the AHRS flight guidance electronics (replaces the C-14D gyros), but the AC electrical system (inverters) remain unchanged (Chapter 2, “Electrical Power Systems”). In the event of a dual inverter failure, a third inverter is installed to provide emergency AC power for the pilot primary flight instruments. This inverter is normally not active unless the primary inverters fail. During the preflight test of the inverters, selecting the inverter test switch to INV 1 TEST or INV 2 TEST causes a standby inverter annunciator for the third inverter to illuminate, indicating it is functioning properly.



INSTRUMENT PANEL ANNUNCIATORS (EFIS)

Annunciator warning lights associated with the integrated flight control system are located directly above the pilot and copilot EADIs on the instrument panels. They are as follows:

FMS1 (or 2) SX—Parallel track. Illuminates when FMS 1 or FMS 2 has been programmed for course SX guidance. Course guidance with respect to a course offset from, but parallel to, the leg shown on the CDU.

VLF1 (or 2) Battery—VLF/Omega battery. Illuminates when the GNS-X RPU is being electrically powered by its own internal stand-by battery. Illuminates during periods of aircraft power losses.

COMPRTR—Each symbol generator for the pilot and copilot EFIS monitors its various raw data inputs. When the input data for the same type data exceed certain parameters, the annunciations illuminate. A symbol for the parameter exceeded is displayed on the EADI as follows:

PTT—Pitch attitude exceeds $\pm 6^\circ$

ROL—Roll attitude exceeds $\pm 6^\circ$.

HDG—Heading data exceeds $\pm 6^\circ$.

LOC—Localizer signals exceed 1/2 dot deviation

GS—Glideslope signals exceed 2/3 dot deviation

ATT—Pitch and roll attitude exceeds $\pm 6^\circ$

ILS—Localizer +40 mV; Glideslope ± 50 mV

AP OFF—Illuminates when either the pilot or the FGS disengages the autopilot. A one-second tone also sounds.

YD OFF—Illuminates when either the pilot or the FGS disengages the autopilot or the YD. The autopilot may be disengaged separately without disengaging the yaw damper by depressing the primary trim switch or the go-around button.

EFIS FAN—EFIS cooling fan. Illuminates if the pilot or copilot EFIS cooling fans or nose avionics bay cooling fans fail.

AHRS AUX PWR—AHRS auxiliary battery pack. AHRS 1 or 2 has transferred to the auxiliary battery due to loss of DC power. Auxiliary battery power is limited to five minutes.

Annunciator warning lights located on the center instrument panel are as follows:

FAIL A RESET—Left flight guidance system (FGS) computer invalid. Can be reset.

FAIL B RESET—Right flight guidance system (FGS) computer invalid. Can be reset.

AIL TRIM L-R—FGS senses a steady-state load on the roll servo. Trim ailerons in the direction indicated.

ELEV TRIM UP-DN—FGS senses a steady-state load on the pitch servo. If this light does not clear automatically within a few seconds, autopilot should be disengaged.

Annunciator lights located on the center pedestal forward of the throttles are as follows:

AHRS BASIC 1-2—True airspeed from one or both DADCs becomes invalid to the No. 1 and/or No. 2 AHRS. AHRS operation in the BASIC mode results in the attitude system (VG) subject to conventional drift and acceleration errors.

AFCS A ON—Left flight guidance computer (A-channel) is being used as the controlling system.

AFCS B ON—Right flight guidance computer (B-channel) is being used as the controlling system. The switchlights may be depressed to enable the pilot to select the controlling system, left or right.

MFD FAN—MFD cooling fan. Illuminates if the fan fails.



FLIGHT GUIDANCE SYSTEM: PRIMARILY SNs 0200—0202 AND 0207—0241

GENERAL

The standard flight instrumentation consists of the SPZ-650 Integrated Automatic Flight Control System (AFCS). The AFCS consists of two independent electronic flight instrument systems, one independently controlled system on each (pilot and copilot) instrument panel (Figure 16-41). Each system (pilot and copilot) may be used in conjunction with dual flight

director mode selectors and the single remote instrument controller to fly the aircraft. The crew may use the system to fly the aircraft manually using the flight director, or automatically using the autopilot controller. The system meets Category II equipment requirements when flown with the autopilot engaged.

SPZ-650 integrated AFCS and EFIS includes dual FZ-500 Flight Directors, EDZ-605 Electronic Flight Instrument System (EFIS), the ADZ-810 Air Data System (pilot side only, unless incorporating SB650-34-105), and a PC-500 Autopilot.

The SPZ-650 AFCS is a complete automatic flight control system. Flight path commands



Figure 16-41. SPZ-650 IFCS Flight Deck: Primarily SNs 0200—0202 and 0207—0241



are generated by the FZ-500 Flight Director Computers, which integrate attitude and heading from the C-14D directional gyros and VG-14A vertical gyros, air data from the ADZ-810 system, and navigation from various navigation components and the EFIS, all via an avionics standard communications bus (ASCB).

The SPZ-650 AFCS requires primarily 28 VDC power for operation. The air data computer, altimeters (copilot side only), and Mach air-speed display require 26 VAC. The weather radar system also requires 26 VAC for antenna stabilization.

FLIGHT DIRECTOR MODES

The flight director mode selector provides all mode selection (except go-around), which is initiated by a remote switch on the throttle(s) for the flight director (Figure 16-42). The flight director command bars on the EADI provide integrated pitch and roll guidance to satisfy the selected mode. The top row of annunciator buttons contains the lateral modes, and the bottom row contains the vertical modes.

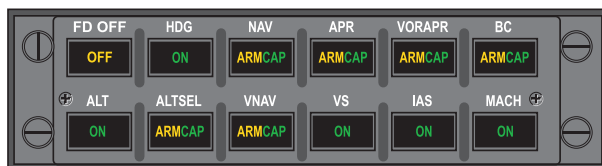


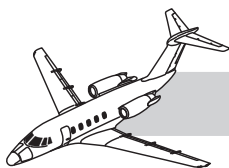
Figure 16-42. Flight Director Mode Selector

The split-light buttons illuminate amber for armed conditions and green for captured. Mode status is also displayed on the respective EADI when applicable. If more than one lateral or vertical mode is selected, the flight director system automatically captures the submode. The flight director mode selector buttons and their function are as follows:

- **HDG (heading)** —Commands the flight director computer to follow the inputs of the heading bug on the EHSI. The command bars on the EADI are then driven to track the position of the heading bug.

While in the heading mode, a lower bank limit can be selected with the BANK LIMIT button on the autopilot controller.

- **FD OFF (flight director off)**—Removes the command bars from view, but the modes from the flight director computer remain engaged.
- **NAV (navigation)**—Allows the flight director computer to arm, capture, and track the selected navigation signal sources (VOR, LOC, AZ, LNAV). When APR is selected, the NAV select button annunciates lateral tracking status
- **APR (approach)**—Selects the appropriate gains to arm and capture the lateral deviation signal for LOC and MLS azimuth, and both lateral and vertical navigation signals for ILS and MLS to meet approach criteria. For localizer or azimuth-only approaches, when APR is selected, only the NAV button annunciates the arm or capture condition.
- **VORAPR (VOR approach)**—Selects the appropriate gains for arm and capture of a VOR course for approach. This mode is normally used within 20 nm of a VOR station. It works identically to NAV mode.
- **BC (back course)**—Commands the flight director computer to track the localizer back course.
- **ALT (altitude)**—Commands the system to hold the current altitude. Capturing the altitude displayed on the VNAV controller allows the system to maintain that altitude.
- **ALTSEL (altitude select)**—Allows the flight director computer to arm and capture an altitude selected with the VNAV controller.
- **VNAV (vertical navigation)**—Allows the system to arm and capture a VOR/DME-based vertical profile, allowing coupled climb or descent to a waypoint altitude. The vertical profile data is entered with the VNAV controller.



NOTE

The VNAV mode is not operational when the Honeywell FMZ-800/900 Flight Management System is the selected navigation source.

- **VS (vertical speed)**—Selects the system to maintain the current vertical speed and to allow a new vertical speed to be selected (with either the autopilot pitch wheel or by using the TCS button) and maintained. The vertical speed target is displayed on the EADI.
- **IAS (indicated airspeed)**—Commands the system to maintain the current indicated airspeed or to allow a new indicated airspeed to be selected (with either the autopilot pitch wheel or by using the TCS button) and maintained. The indicated airspeed target is displayed on the EADI.
- **MACH**—Works identically to IAS mode except it commands the flight director to maintain the current Mach value. The target value is displayed on the EADI.

AUTOPILOT

Controller

The PC-500 Autopilot Controller (Figure 16-43) provides the means of engaging the autopilot and yaw damper, as well as manually controlling the autopilot through the TURN

knob and PITCH wheel. Whenever the autopilot is engaged, it will fly basic roll and pitch hold or the selected lateral or vertical flight director modes. Controls and indicators on the autopilot controller are as follows:

- The AP engage button engages the autopilot and yaw damper simultaneously (cannot be used to disengage the autopilot). Engagement is annunciated by the illumination of the button. The autopilot may be engaged with the aircraft in any reasonable attitude. With no flight director modes selected, the autopilot rolls to wings level upon engagement and holds the existing pitch attitude and aircraft heading. If any flight director modes are active prior to engagement, then the autopilot automatically couples to the flight director modes. The autopilot will not engage unless the turn knob is centered.
- The YD engage button only engages the yaw damper. (It will not disengage the autopilot.)
- A sustained request for elevator trim is indicated by illumination of UP or DN. This annunciation remains as long as the elevator servo is required to hold a force to maintain desired aircraft attitude. Both indicators should be off prior to engaging the autopilot.
- Selection of the bank limit mode on the autopilot controller provides a lower maximum bank angle (17°) while in the flight director heading select mode. LOW illu-

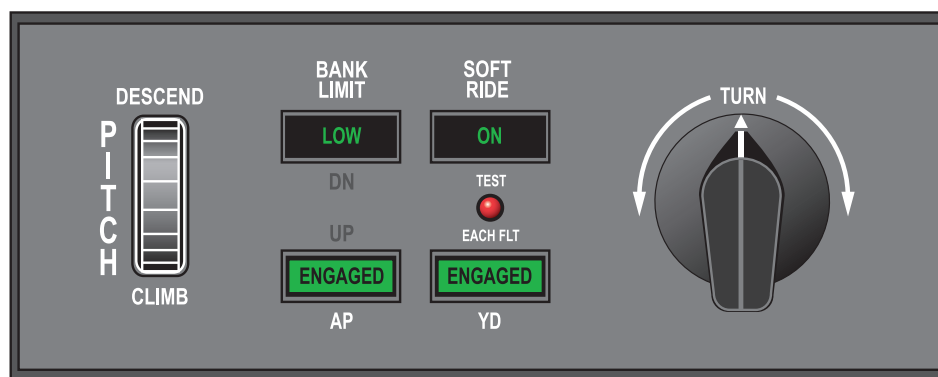


Figure 16-43. Autopilot Controller



minates on the BANK LIMIT switch. The lower bank limit is inhibited and LOW is extinguished during NAV mode captures. If heading select is again engaged, BANK LIMIT is again illuminated. Pressing BANK LIMIT when illuminated returns the autopilot to normal bank limits.

- SOFT RIDE provides reduced pitch and roll autopilot gains while still maintaining stability in rough air. This mode may be used with any flight director mode selected.
- Rotation of the TURN knob out of detent results in a roll command. The roll angle is proportional to and in the direction of the turn knob rotation. The turn knob must be in detent (center position) before the autopilot can be engaged. Rotation of the turn knob cancels any other previously selected flight director lateral mode.
- Rotation of the PITCH wheel with the autopilot engaged results in a change of pitch attitude proportional to the rotation of the pitch wheel and in the direction of wheel movement. Movement of the pitch wheel cancels only ALT HOLD or ALTSEL CAP. With vertical modes of VS, IAS, or MACH selected on the mode selector, rotation of the pitch wheel changes the respective EFIS displayed vertical mode reference (with the autopilot engaged). VS, IAS, or MACH mode may be canceled by pressing the mode button on the mode selector. If the air data modes are not selected, the pitch wheel works as described above. However, for safety, movement of the pitch wheel has no effect with the autopilot coupled to the glideslope or glidepath.

Preflight Test

- Normal elevator trim must be activated prior to starting the autopilot test. Otherwise, the test cannot be completed.
- Failure to complete and pass this autopilot test inhibits the autopilot from engaging. The test can be performed either on the ground or when airborne.

- The vertical and directional gyro flags must be out of view.

Autopilot Test

1. Check that all AFCS circuit breakers are in.
2. Press and hold the TEST EACH FLT button on the autopilot controller.
 - a. Observe that the AP TORQUE annunciator is illuminated (Figure 16-44).
 - b. Observe that after three seconds the AUTOPILOT OFF annunciator is illuminated and that the horn sounds (one second).
3. Release the test button. The test is complete.

NOTE

If during the flight there is a power interruption to the AFCS, this test must be performed again prior to re-engaging the autopilot.

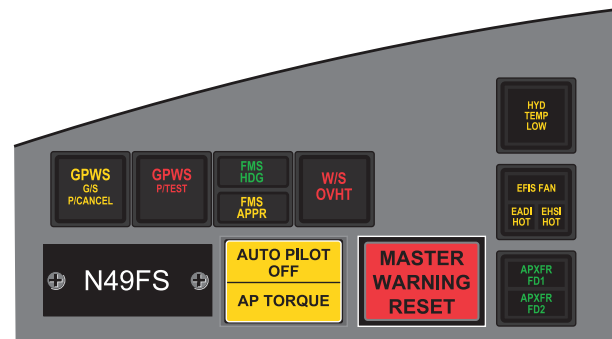
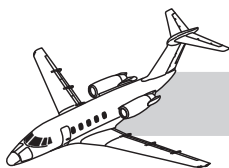


Figure 16-44. Instrument Panel Annunciators

Annunciators

- AP TORQUE illuminates when the autopilot pitch or roll serve torque limit has not switched to reduced gain (Figure 16-44). If the annunciator remains illuminated, autopilot use is prohibited.
- AUTOPILOT OFF indicates that the autopilot has been manually disengaged or has automatically disengaged due to an internal malfunction.



Remote Switches

Autopilot Transfer

The autopilot transfer switch (AP XFR FD 1–FD 2) selects which flight director computer provides signals to the autopilot computer for control of the servos (Figure 16-44). It also selects which side of the cockpit can use the pitch wheel to command the value of VS, IAS, and Mach for flight reference.

AP ENGAGE annunciators are located on the upper instrument panels, one directly above each EADI. Below each annunciator is an arrow which illuminates in green, pointing in the direction of the controlling flight director coupled to the autopilot.

Autopilot Disconnect

To simultaneously disconnect the autopilot and yaw damper push the red AP/TRIM/NWS DISC button on the control wheel. The autopilot can also be disengaged by performing one of the following actions:

- Pressing the go-around button on the throttles (provided the flight director is engaged)
- Activating primary pitch trim (yaw damper remains engaged)

Touch Control Steering (TCS) Switches

Also mounted on the control wheels, the TCS switches allow the pilots to manually change aircraft attitude, altitude, airspeed, and/or vertical speed without disengaging the autopilot.

Go-Around Switches (Throttle)

The go-around switches are located on the throttle (Figure 16-45). They will disengage the autopilot and command a wings-level 10° nose up attitude.

Vertical Gyro Fast-Erect Switch

This remote-mounted momentary switch provides for manual erection of the vertical gyro and is spring-loaded in the NORM position



Figure 16-45. Throttle Go-Around Switch

(Figures 16-46 and 16-47). The switch should be depressed and held in the HI position to erect the gyro only when the aircraft is in stable level flight. When the switch is in the HI position and the gyro is up to speed, the gyro will erect at about 20° per minute. Pitch axis movement will be seen only when the switch is released.

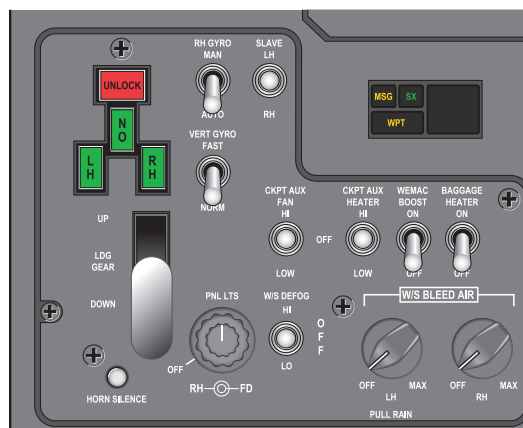


Figure 16-46. Copilot Lower Switch Panel

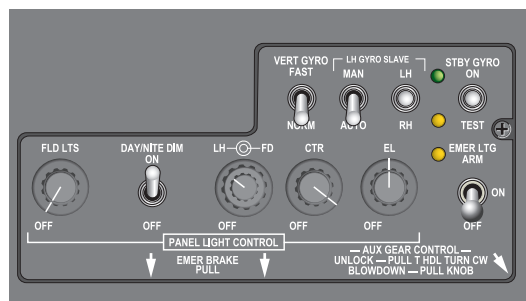
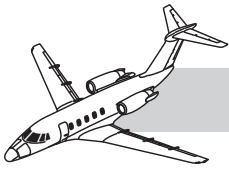


Figure 16-47. Pilot Lower Switch Panel



Directional Gyro AUTO-MAN Switch

This remotely located switch is used to select either the slaved or free gyro mode of operation.

Slave (LH and RH) Switch

If needed during compass initialization, this remotely located switch manually positions the heading dial for compass synchronization in the slave mode and for gyro updating in the free mode.

Emergency Descent Mode

This mode is automatically initiated by an input from a cabin pressure altitude sensor. The parameters for this mode to actuate are as follows:

- Aircraft altitude of 34,275 feet or higher
- Cabin altitude or 13,500 feet
- The autopilot engaged

This mode disengages any previously coupled flight director modes. The red EMERG DESCENT annunciator panel light illuminates, and the MASTER WARNING RESET switch-lights flash. The autopilot initiates a pitch-down to capture and maintain $V_{MO} - 10$ KIAS and rolls left to a 35° bank angle for approximately 48 seconds to clear enroute airways.

The autopilot must be disengaged to cancel the emergency descent mode before reselecting normal autopilot/flight director operation. If the autopilot remains engaged during the emergency descent mode, it commands a flare to level at 15,000 feet.

ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

General

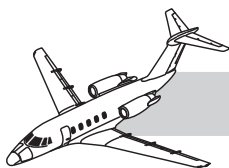
The standard flight instrument configuration is the EDZ-605 EFIS, which consists of dual-tube Honeywell ED-600 Electronic Displays on the pilot and copilot instrument panels. These cathode ray tubes are identical and interchangeable (Figure 16-48). The top tubes are used for electronic attitude director indi-

cators (EADIs) and have conventional inclinometers attached to the lower bezel of the display. The lower tubes are used as electronic horizontal situation indicators (EHSIs). The optional EDZ-805 EFIS system may be installed, which consists of ED-800 tubes which are larger displays.

Associated with each display system (pilot and copilot) is a symbol generator and a display controller (DC-811) (Figure 16-49). The heart of the EFIS system is the symbol generators, which receive and process all aircraft sensor



Figure 16-48. ED-600 EFIS Displays



inputs. The data is then transmitted to the EFIS display tubes. The pilots control display formatting with the display controllers. A common instrument remote controller, located on the center pedestal directly below the pilot display controller interfaces with the symbol generators to provide remote heading and course selection.

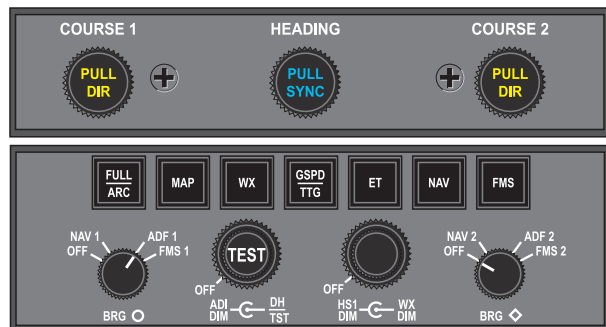


Figure 16-49. RI-206S Instrument Remote Controller and DC-811 Display Controller

The EFIS system provides for display integration, flexibility, and redundancy. The flight crew has essential display information, navigation, air data commands, attitude, and caution/failure warning systems integrated into their prime viewing areas. Each symbol generator is capable of driving four ED-600/800 Electronic Displays. If one side symbol generator fails, the remaining symbol generator will drive the displays on both sides.

Composite Format

If one ED-600 display tube fails, a composite attitude/heading display format can be displayed on the remaining display tube on that side. This is accomplished by turning the ADI or HSI dim control OFF on the display controller of the failed display tube.

Reversionary Mode

EFIS reversionary select buttons are located on the left side of the pilot instrument panel and on the copilot reversionary switch panel. The reversionary select buttons allow the pilots to select alternate sources of symbol generators,

heading, and attitude. When alternate sources are selected, one source driving both sides, the source annunciator on the display tubes will be amber, i.e., MAG 1 or 2, and/or SG 1 or 2.

Multifunction Display System (MFD) Optional

An optional MDZ-605 Multifunction Display System (MFD) is installed on the lower center instrument panel. The MFD display serves as a radar indicator or as a backup to the EFIS system. The MFD symbol generator can be used to back up the EFIS symbol generators. The MFD display tube can also be used to back up the EHSI display tubes. The MFD system expands on the navigation mapping capability of the EFIS system.

The MFD system is controlled by an MFD controller installed on the center pedestal. The controller is used to select various modes of operation: MAP, PLAN, weather, checklist (normal or emergency), and EFIS backup modes. Additional joysticks are installed on both the pilot and copilot forward side console armrests to operate the checklist mode.

NOTE

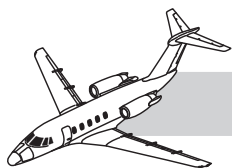
Consult the Honeywell SPZ-650 pilot manuals for the Citation VI and Section III, "Instrumentation and Avionics," of the *Airplane Operating Manual* for detailed operation instructions for the flight guidance system.

C-14D COMPASS SYSTEMS

General

The pilot EHSI, the copilot RMI, and the pilot flight director are driven by the pilot No. 1 C-14D slaved gyro system. The system consists of a directional gyro, a flux detector, a mode selector switch, a remote compensator, and a slaving indicator on the EHSI. The pilot system is powered by the main DC electrical system.

The LH GYRO SLAVE switch located on the lower left switch panel has two positions, labeled "MAN" and "AUTO" and allows the



compass to be operated in the slaved or free DG mode. In the AUTO (slaved) mode, the compasses align at approximately 3 to 5° per second. When MAN is selected, the EHSI and the copilot compass card can be moved left or right at a rate of 30° per minute by toggling the LH–RH switch. In this mode, the slaving indicator on the EHSI disappears. Under normal operating conditions, the gyros remain in the AUTO (slaved) mode.

The copilot C-14D compass system is identical to the pilot system. The copilot system drives the right-side EHSI, the pilot RMI, and the copilot flight director. The copilot C-14D compass system is powered from the emergency DC bus (Chapter 2, “Electrical Power System,” of this manual for emergency operating procedures.)

Radio Magnetic Indicators (RMIs)

Dual RMI-36 radio magnetic indicators are mounted on both instrument panels. ADF and VOR magnetic bearing information is displayed on each RMI. The single-bar bearing pointers display VOR 1 or ADF 1. The double-bar bearing pointers display VOR 2 or ADF 2. Push-type selector switches for each pointer are mounted on the lower case of the RMI. The compass card for each RMI is driven by the opposite-side C-14D directional gyro.

Emergency Power Conditions

Operating on emergency DC electrical power only, the pilot RMI is operational (driven by the No. 2 C-14D). During a loss of both inverters (normal DC power still available), the pilot RMI compass card and No. 1 needle is operational, provided the FD No. 1 circuit breaker (DC) on the copilot circuit breaker panel is pulled.

AIR DATA SYSTEM (ADZ-810)

General

The ADZ-810 Air Data System includes the following equipment:

- AZ-810 Air Data Computer(s)
- Dual SI-225S Mach/Airspeed Indicators
- VN-800 VNAV Altitude Controller
- DS-125A TAS/TAT/SAT Indicator
- Altimeters

The AZ-810 receives both digital and analog inputs, processes this data, and supplies both digital and analog outputs. The AZ-810 receives pitot-static pressures and total air temperature inputs for computing standard air data functions. The AZ-810 provides data for the pilot altimeter, Mach/airspeed indicators, transponders, EFIS/MFD symbol generators, flight directors, optional flight data recorder, and other components of the flight control system. The DS-125A TAS/TAT/SAT Indicator displays true airspeed, total air temperature, and selectable static air temperature.

SI-225S Mach/Airspeed Indicators

The pilot SI-225S Mach/Airspeed Indicator is a servoed instrument featuring a Mach counter display and an IAS pointer display (Figure 16-50). The maximum allowable V_{MO} and M_{MO} speeds are computed by the air data computer. A maximum allowable airspeed (barber pole) pointer will move to indicate maximum airspeed with respect to altitude. The indicator is fully electric (26 VAC) and displays OFF flags during periods of power failure.

The copilot airspeed indicator (Figure 16-51) is operated by the copilot pitot-static system directly. The air data computer provides the Mach portion to the copilot indicator. Consequently, if electrical power is lost, the copilot indicator will still display KIAS information. If SB650-34-105 is incorporated, the copilot airspeed indicator may be supplied information by the copilot air data computer, in this case, a red OFF flag will be displayed in the event a power failure occurs.

Four movable indices (bugs) of different colors are located on the bezel glass ring for placement to critical airspeeds.

**Figure 16-50. Pilot Airspeed Indicator****Figure 16-52. Pilot Altimeter****Figure 16-51. Copilot Airspeed Indicator**

Altimeters

Pilot Altimeter

The pilot encoding altimeter system is driven from the digital air data computer (Figure 16-52). It is an electrically servoed counter/pointer display of barometrically corrected altitude.

Copilot Altimeter

The copilot altimeter is a conventional barometric type with a counter/pointer readout. An optional encoding altimeter, which provides an alternate source of uncorrected barometric altitude information for transponders, is available for the copilot. In installations having the optional copilot encoding altimeter,

a switch is provided on the instrument panel to control selection of the source of altitude information which is transmitted to the transponders. The copilot altimeter has a dedicated static system which operates from the forward left and forward right static ports. If SB650-34-105 is incorporated, the copilot altimeter will be supplied information by the copilot air data computer. In this case, a red OFF flag will be displayed in the event a power failure occurs.

TAS/TAT/SAT Indicator

The DS-125A TAS/TAT/SAT Indicator provides a digital display of both true airspeed (TAS) and total air temperature (TAT) or static air temperature (SAT) as generated by the digital air data computer (DADC). The indicator normally displays true airspeed and total air temperature. Pressing the SAT button changes the temperature display to static air temperature as the button is held.

VN-800 VNAV Altitude Controller

The VNAV computer/controller provides the data inputs for the altitude preselect mode (ALTSEL), altitude alert, and vertical navigation (VNAV) mode (Figure 16-53). Data is entered into the computer by rotating the data select switch to the desired position and then setting the required value with the data set



knob. The profile mode is selected and annunciated by the green illuminated PROFILE select switch. It is a push-on/push-off switch which is illuminated when the VNAV profile mode is selected. The VNAV operates only when using NAV 1 with the DME set to NAV 1 and locked on the selected VORTAC (not in HOLD). VNAV information is valid only when flying directly TO or FROM a VORTAC.



Figure 16-53. VN-800 Vertical Navigation Controller

Vertical Speed Indicators

Two instantaneous vertical speed indicators indicate vertical velocity from 0 to 6,000 feet per minute, either up or down (Figure 16-54). Their operation differs from conventional vertical speed indicators in that there is zero time lag between aircraft displacement and instrument indication. Accelerometers sense any change in normal acceleration and displace the needle before an actual pressure change occurs. The instruments are pneumatic.

STANDBY FLIGHT INSTRUMENTS

Standby Attitude Gyro

A standby attitude indicator is located on the pilot instrument panel. It normally operates on emergency DC electrical power through the STBY ATT IND circuit breaker on the right circuit breaker panel. Power to the gyro is controlled by the STBY GYRO switch, with ON, OFF, and TEST positions, located on the pilot lower instrument panel. An emergency battery pack in the nose avionics compartment



ANALOG VSI

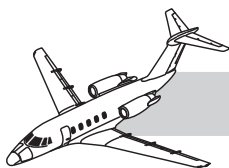


ELECTRONIC VSI

Figure 16-54. Vertical Speed Indicators

is an emergency source of power for the standby gyro if main DC bus voltage falls below minimum. This is indicated by an amber power-on light adjacent to the STBY GYRO switch, provided the switch is in the ON position. The battery pack also provides power for emergency instrument lighting for the pilot primary flight instruments.

The battery pack is continuously charged by the emergency DC electrical system and should be fully charged in the event of an electrical power failure. The STBY GYRO power switch must be in the ON position for automatic transfer to emergency battery pack power. The gyro will operate for a minimum of 30 minutes on emergency battery pack power. The amber light adjacent to the switch will illuminate when the gyro is operating on its own power source (standby).



battery power). When the switch is held to the TEST position, a self-test of the emergency battery pack and associated electrical circuits is accomplished. The amber light adjacent to the switch illuminates if the test is satisfactory and the battery pack is fully charged. The green light illuminates with the switch ON and aircraft power operating the standby gyro and charging the emergency battery pack.

The standby gyro is caged by pulling the PULL TO CAGE knob and rotating it clockwise.

CAUTION

When uncaging, do not release the PULL TO CAGE knob suddenly so that it snaps back; this can damage the gyro.

FLIGHT GUIDANCE SYSTEM: PRIMARILY SNs 0001 — 0178

GENERAL

The flight instrumentation consists of the standard SPZ-650 autopilot/flight director instrument system. One independently controlled system on each of the pilot and copilot instrument panels (Figure 16-55) is the most common option. Each system (pilot and copilot) is controlled by individual flight director mode selectors and instrument remote controllers located on the center pedestal (Figure 16-56 and 16-58). The crew may use the system to fly the aircraft manually using the flight directors, or automatically using the autopilot.



Figure 16-55. Flight Guidance System: Primarily SNs 0001 — 0178 (Standard)



Figure 16-56. Standard Flight Director Mode Selector

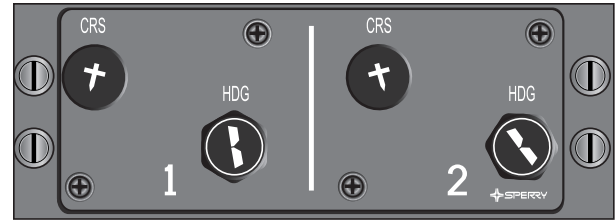


Figure 16-58. Remote Instrument Controller



Figure 16-57. Standard Flight Director System

The SPZ-650 autopilot/flight director system is a complete automatic flight control system which provides flight director, automatic pilot, flight instrumentation (including gyro references, radio altitude, and flight instruments), and a digital air data system including air data displays, altitude alerting, vertical navigation (VNAV), and altitude reporting.

The system requires 28-VDC and 26-VAC/115 VAC power for operation (Chapter 2, “Electrical Power Systems”).

SPI-501 FLIGHT DIRECTOR SYSTEM

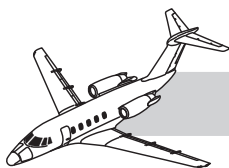
General

The Sperry SPI-501 Flight Director system includes a full complement of horizontal and vertical flight guidance modes. These include all radio guidance modes, INS, RNAV, VNAV, and air-data-oriented vertical modes.

The system features AD-650 series ADIs and RD-650 HSIIs, which are normally the full 5-inch indicators (Figure 16-57).

Attitude Director Indicators (ADIs)

The AD-650 Attitude Director Indicator (ADI) displays pitch and roll attitude (sphere), flight director commands (single-cue command bar), glideslope, localizer deviation, radio altitude, rate of turn, speed command, appropriate fail flags, inclinometer, rising runway, self-test switches, and mode annunciation (Figure 16-57).



ATT Test

When depressed, the sphere will show an approximate attitude change of 20° of right bank and 10° pitch up. The ATT warning flag will appear. In addition, the FD warning flag and all mode annunciators except DH illuminate.

CAUTION

Do not leave the ATT test button depressed for longer than 20 seconds.

Horizontal Situation Indicators (HSIs)

The RD-650 Integrated Horizontal Situation Indicator (HSI) displays heading, course deviation (CDI), vertical deviation, ADF or VOR bearings, selected heading (HDG bug), selected course with digital readout, digital distance display (DME), waypoint alert light, and appropriate annunciators and flags. It also provides heading and course error signals to the flight director computer (Figure 16-57).

Remote Instrument Controllers

The remote heading and course select controllers are located on the center pedestal (Figure 16-58) and drive the HSI heading bugs and the HSI course pointers (digital readout).

FZ-500 Flight Director Computer

The flight director computer contains all computation and mode selection electronics for flight path modes.

MS-500A Flight Director Mode Selector

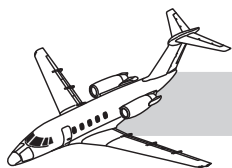
The MS-500A provides all mode selections (except go-around which is initiated by a remote switch) for the flight director (Figure 16-56 or 16-59). The top row of push-button annunciators contains the lateral modes, and the bottom row contains the vertical modes. The split-light pushbuttons illuminate amber for armed conditions and green for captured. Mode engagement is annunciated by the illumination of the pushbuttons on the mode selector. Disengagement of each mode is accomplished by repushing the mode pushbutton (push-on, push-off) or by selection of another pitch or roll mode. A roll mode must be selected in order for the command cue to be in view. If a roll mode but no pitch mode is selected, the pitch command will be pitch hold. The go-around mode is selected by pressing the go-around pushbutton on the throttle(s).

Navigation Select Switching (NAV SEL)

The NAV SEL button on the mode selector is used to select whether No. 1 or No. 2 NAV receiver supplies inputs to the flight director. The number illuminated in the mode selector button indicates which NAV receiver is being used. The selection between No. 1 and No. 2 NAV receiver alternates each time the button is pressed. In dual flight director installations, selection of NAV source (NAV 1 or 2) will force a change to the opposite NAV source in the other flight director mode selector. It is not possible to power both flight directors from the same NAV receiver. Course and heading for the No. 1 flight director are always provided by the No. 1 HSI and course and heading for the No. 2 flight director are always provided by the No. 2 HSI.



Figure 16-59. MS-500A Flight Director Mode Selectors

**CAUTION**

Caution in the use of the NAV SEL switch during autopilot and flight director operation is advisable, since inadvertent activation of the switch may result in an undesirable autopilot/flight director reaction.

Flight Director Mode Annunciation

The following is a listing of all flight director mode annunciations along the top face of the ADI:

HDG—Heading select mode is engaged.

NAV—A radio navigation mode (VOR, RNAV) has been captured and is being tracked.

LOC—Illuminated when localizer is captured.

APR—Illuminated at VOR approach capture or localizer capture.

GS—Illuminated when glideslope captured.

VRT—The system is in vertical speed hold or altitude preselect capture.

ALT—Altitude hold mode is engaged.

BC—Illuminated when the backcourse localizer is captured.

SPD—The system is in airspeed hold or Mach hold mode.

VN—VNAV capture has occurred.

DH—Illuminated when aircraft reaches the preset decision height.

GA—Go-around is selected.

For detailed instructions, consult the Sperry pilot manual for the Citation III and Section III of the *Operating Manual*.

RG-203 Rate Gyro

The rate gyro drives the rate-of-turn display on the ADI (Figure 16-57).

Comparator Monitor

Comparator warning lights located on the upper portion of each instrument panel (pilot and copilot) illuminate if the ADIs display an attitude difference out of tolerance (Figure 16-60). VG ROLL and/or VG PITCH illuminates if the roll and/or pitch axis are more than 6° difference between the two ADIs.



Figure 16-60. ADI Comparator Monitor Annunciator

SPZ-650 AUTOPILOT SYSTEM**General**

The SPZ-650 is a torque-programmed autopilot system which provides full three-axis control. The autopilot includes synchronization engage logic functions and pitch and roll comparator monitoring and provides proper coupling of flight director commands or manual controller inputs for all modes.

SP-650 Autopilot Computer

The autopilot computer contains all autopilot electronics and servo amplifiers.

PC-500 Autopilot Controller

The controller contains the autopilot engage, yaw damper engage, bank limit, and soft ride pushbuttons, TURN knob, PITCH wheel, trim indicator, and autopilot press-to-test switch (Figure 16-61).

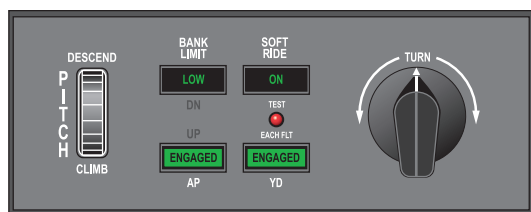
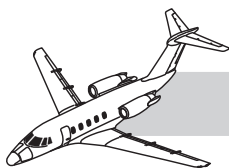


Figure 16-61. Autopilot Controller

Autopilot Engage (AP) Pushbutton

The AP engage pushbutton engages the autopilot and yaw damper when pressed. ENGAGE will illuminate in both buttons. The autopilot can be engaged in any reasonable flight attitude; however, the primary pitch trim switch must be momentarily activated to allow re-engagement if the autopilot has been engaged and then disconnected.

If a flight director lateral or pitch mode is not selected upon or after autopilot engagement, pitch hold and heading hold modes will be engaged. Neither of these modes is annunciated. If a flight director mode is engaged, engaging the autopilot will automatically couple the autopilot to the flight director.

Yaw Damper Engage (YD) Pushbutton

The yaw damper may be engaged without engagement of the autopilot by pressing the YD engage button. ENGAGE will illuminate in the button. The yaw damper may be used at all times during flight except takeoff and landing. Passenger comfort is increased by its use, and aircraft control is greatly enhanced in the roll mode at higher altitudes. During single-engine operation, the yaw damper should be disengaged, the aircraft trimmed for straight-ahead flight, and the yaw damper re-engaged.

Test (TEST EACH FLT) Button

The autopilot will not engage unless its current monitor circuit is tested after each power-up and found to be operational. To test the autopilot, accomplish the following:

1. Apply electrical power to the aircraft.
2. Prearm the stabilizer trim by momentarily actuating either the pilot or copilot trim switch in either direction.
3. Press and hold the TEST EACH FLT button on the autopilot controller. The AP TORQUE annunciator illuminates immediately (Figure 16-62).
4. After approximately three seconds, the AUTOPILOT OFF warning annunciator on the instrument panel momentarily illuminates and the autopilot-off horn sound. This indicates that the test results are positive, and the autopilot can be engaged and disengaged as needed for the flight.
5. Engage the autopilot, and verify proper response from the control column and wheel during pitch wheel command and turn knob command. Disengage the autopilot by the AP/TRIM/NWS DISC switch. Verify that the AUTOPILOT OFF light illuminates and the horn sounds.

NOTE

The test must be accomplished any time that power to the autopilot has been off or interrupted. The preflight test can be accomplished on the ground or during flight.

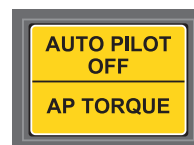
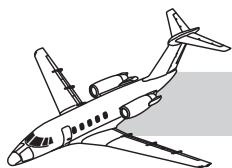


Figure 16-62. AUTOPILOT OFF/AP TORQUE Annunciator

Autopilot Disengage

To simultaneously disconnect the autopilot and yaw damper press the red AP/TRIM/NSW DISC pushbutton located on the control wheel. Insertion of manual electric elevator trim will also disengage the autopilot.



The autopilot may be disengaged by the following:

- Selection of go-around mode (yaw damper remains engaged).
- Use of manual electric elevator trim (yaw damper remains engaged).
- Actuation of compass LH–RH switch.
- Actuation of vertical gyro FAST ERECT switch (HI position).
- Pressing the autopilot TEST EACH FLT button on the autopilot controller.
- Pulling the AP, AC or DC circuit breaker.

The following malfunctions cause the autopilot to automatically disengage:

- Vertical gyro failure
- Directional gyro failure
- Autopilot power or circuit failure
- Torque-limiter failure (AP TORQUE annunciator on). Do not attempt re-engagement of the autopilot

Disengaging under any of the above four conditions illuminates the AP OFF light. Pressing the autopilot disengage switch turns off the light. The autopilot-off warning horn sounds for one second at disengagement.

PITCH Wheel

With the autopilot engaged, rotation of the PITCH wheel results in a change of pitch attitude proportional to the rotation of the wheel and in the direction of wheel movement. With the autopilot engaged and coupled to a flight director pitch path mode (ALT, ALTSEL CAP, IAS, VS, MACH, or VNAV CAP), movement of the PITCH wheel resets the pitch mode, and both the flight director and autopilot revert to the pitch hold mode. Movement of the PITCH wheel with the autopilot coupled to the glideslope has no effect.

TURN Knob

With the autopilot engaged, rotation of the TURN knob out of detent results in a roll angle

proportional to and in the direction of the TURN knob rotation up to a maximum bank angle of 30°. The TURN knob must be in the detent (center) position before the autopilot can be engaged. With the autopilot engaged and couple to a flight director roll path mode (HDG, NAV, BC, or VORAPR), movement of the TURN knob out of the detent position cancels the selected roll path mode (ARM or CAP).

BANK LIMIT Pushbutton

Selection of the BANK LIMIT mode on the autopilot controller provides a lower maximum bank angle while in the heading select mode (approximately 17°). LOW illuminates on the BANK LIMIT switch. The lower bank limit is inhibited and LOW is extinguished during NAV mode captures. If heading select is again engaged, the BANK LIMIT switch will again be illuminated. Pressing the BANK LIMIT when illuminated returns the autopilot to normal bank limits.

SOFT RIDE Pushbutton

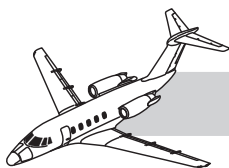
Soft ride reduces autopilot gains while still maintaining stability in rough air. This mode may be used with any flight director mode selected. This mode is not recommended during ILS approaches.

ELEV TRIM Indicator

The elevator trim indicator shows out-of-trim condition by displaying UP or DN when a sustained signal is being applied to the elevator servo. The indicator should be off before the autopilot is engaged.

Operation of Flight Director Modes with Autopilot Engaged

Whenever the autopilot is engaged, it flies the roll and/or pitch mode selected on the flight director/autopilot mode selector. When the autopilot is engaged and a roll mode is selected, operation of the turn knob cancels the roll mode, biasing the command cue from view. When the autopilot is engaged with a pitch mode selected, operation of the pitch wheel cancels the pitch mode, biasing the command



cue from view of no roll mode is selected, or the system to the pitch sync mode if a roll mode is on. Moving the pitch wheel syncs the flight director command.

Touch Control Steering (TCS)

Touch control steering (TCS) enables the aircraft to be maneuvered manually during autopilot operation without cancellation of the autopilot or any selected flight director modes. To use touch control steering, press the TCS button (the autopilot temporarily disengages), maneuver the aircraft manually, and release the TCS button (the autopilot re-engages). TCS is operable with all autopilot modes. If the autopilot is engaged in a bank and it is desired to hold the bank, press the TCS button, engage the autopilot, and release the TCS button. The bank will be maintained if it is in excess of 6°. The aircraft can be rolled level with the turn knob. The memory function holding the autopilot in a bank is canceled when the turn knob is moved out of the detent.

Whenever a roll mode is selected without a pitch mode, the command cue displays a pitch attitude hold command. The pitch attitude can be changed by pressing the TCS button on the control wheel and maneuvering the aircraft. The command cue is synchronized to zero (pitch sync) while the button is pressed. Upon release of the button, the pitch command maintains the new pitch attitude. In the pitch hold mode, the command cue is biased out of view if the VG is not valid.

Go-Around Mode

To select go-around mode, press the go-around button on either throttle. GA is annunciated on the ADI, and the command bars display a wings-level pitch-up command of 10°. All other modes are overridden when go-around is selected.

After go-around has been selected, any roll mode can be selected; the wings-level command is then canceled. The go-around mode may be canceled by selecting any other pitch mode, engaging the autopilot, or pressing the touch control steering (TCS) button.

Selecting go-around automatically disengages the autopilot if coupled to the flight director except for the yaw damper. In dual flight director installations, a go-around switch is also located on the right side of the right throttle. Pressing either go-around switch activates the go-around mode in both flight director systems.

Emergency Descent Mode

This mode is automatically actuated by an external logic input from a cabin pressure differential sensor. The parameters for actuating the mode are a cabin altitude of 13,500 feet and aircraft altitude of 34,275 feet. Emergency descent mode clears any previously coupled modes. EMERG DESCENT MODE will be annunciated in red on the aircraft annunciator panel.

The autopilot pitch axis initiates a pitch-down to capture and maintain $V_{MO}-10$ knots speed. This speed is held by the pilot moving the throttles to idle and deploying the speedbrakes and spoilers. The roll axis commands a 35° bank to the left for approximately 48 seconds in order to clear the enroute airway.

An auto-level flare terminates the emergency descent at approximately 12,000 feet for aircraft SNs 0001—0043 not incorporating SB650-22-1 or approximately 15,000 feet for aircraft SNs 0044 and subsequent and aircraft SNs 0001—0043 incorporating SB650-22-1. That altitude is held until pilot action is taken. The autopilot must be disengaged to cancel the emergency descent mode before standard autopilot/flight director operation can be resumed.

Autopilot Transfer

For dual flight director installations, the pilot and copilot attitude and directional gyros can be switched to provide alternate attitude and directional sensing sources for the autopilot. A switchlight marked APXFR FD1 and APXFR FD2 is installed on the pilot upper instrument panel (Figure 16-63). Pressing the switch transfers gyro control of the autopilot, as desired, between the pilot and copilot flight director systems. The system controlling the autopilot is annunciated by illumination of the switch. If FD1 is selected, the pilot C-14D



and VG-14A control the autopilot; conversely FD2 refers to the copilot flight director. In single flight director installations, the autopilot can be controlled only by the pilot vertical gyro and directional gyro.



Figure 16-63. Autopilot Transfer Switchlight

C-14D COMPASS SYSTEMS

Pilot System

The pilot horizontal situation indicator (HSI), the flight director, and the copilot radio magnetic indicator (RMI) are all driven by the pilot C-14D slaved gyro system. The system consists of a directional gyro, a flux detector, two mode selector switches, a dual remote compensator, and a slaving indicator in the HSI. The directional gyro operates from 28 VDC. Two LH GYRO SLAVE switches located on the left switch panel, one marked AUTO-MAN and the other RH-LH, allow selection of automatic (slaved) or manual (unslaved) operation of the pilot C-14D. When the manual position is selected, the HSI compass card can be moved left or right at a rate of 30° per minute by toggling the switch to the RH or LH position. Manual operation gives accurate short-term heading reference when magnetic information is unreliable. Under normal operating conditions, the pilot C-14D GYRO SLAVE switch will be left in the AUTO position. Fast slaving of the pilot C-14D occurs in the AUTO mode at a minimum rate of 30° per minute and continues at that rate until the gyro is slaved to the flux detector. It then continues to maintain slaving at a rate of 2.5 to 5.0° per minute.

The pilot C-14D is normally powered from the avionics bus. In the event of a DC power failure, placing the battery switch to the EMER

position switches the power source to the emergency DC bus to regain C-14D operation.

In EMERGENCY (battery power), the pilot heading indicator will operate along with the course deviation indicator (CDI) on the HSI (26 VAC from the pilot C-14 D gyro static inverter). The HSI bearing pointer, however, will not operate. Both pilot RMIs will be inoperative.

Copilot System

The copilot C-14D s system drives the copilot HSI, the pilot RMI, and when installed, the copilot flight director.

Two GYRO SLAVE switches, marked AUTO-MAN and LH-RH are located on the copilot right panel. Operation of the switches is the same as that described in the pilot C-14D system.

The copilot heading information is provided by a C-14D gyro in both the standard and dual flight director installations.

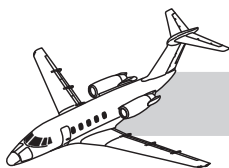
The pilot and copilot C-14D gyros are interchangeable; however, the gyro's static inverter is used only for gyro slave function when installed in the copilot position.

The copilot heading indicator will not operate on EMERGENCY (battery) power.

NOTE

In the event of loss of both inverters, perform the following procedure:

- On SNs 0001—0066, with standard flight directors, place the battery switch to OFF to regain full control of the pilot HSI.
- On SNs 0067—0178 and all EFIS-equipped aircraft, pull the FD 1 DC-powered circuit breaker on the right circuit breaker panel to regain control of the pilot RMI and No. 1 NAV needle.



Radio Magnetic Indicators (RMIs)

Dual RMI-36 radio magnetic indicators are mounted on both instrument panels (Figure 16-64). ADF and VOR magnetic bearing information is displayed on each RMI. The single-bar bearing pointers display VOR 1 or ADF 1. The double-bar bearing pointers display VOR 2 or ADF 2. Push-type selector switches for each pointer are mounted on the lower case of the RMI. The compass card for each RMI is driven by the opposite-side C-14D directional gyro.



Figure 16-64. Radio Magnetic Indicator

ADZ-800 AIR DATA SYSTEM

General

The AZ-800 digital air data computer provides outputs for not only the basic altitude hold and airspeed gain programming functions, but also the more sophisticated vertical speed, airspeed, and Mach control functions to the autopilot and flight director. The AZ-800 also provides outputs to drive the totally servoed altimeter and Mach/Airspeed indicators.

SI-225S Mach/Airspeed Indicator

The SI-225S Mach/airspeed indicator is a servoed instrument featuring a Mach counter display and an IAS pointer display. The maximum

allowable V_{MO} and M_{MO} speeds are computed by the air data computer. A maximum allowable airspeed (barber pole) pointer will move to indicate maximum airspeeds with respect to altitude. The indicator is fully electric (26 VAC) and displays OFF flags during periods of power failure.

The copilot airspeed indicator is operated by the copilot pitot-static system directly. The air data computer provides the Mach portion to the copilot indicator. Consequently, if electrical power is lost, the copilot indicator will still display KIAS information only. If SB650-34-107 is incorporated, the copilot airspeed indicator may be supplied information by the copilot air data computer, in this case, a red OFF flag will be displayed in the event a power failure occurs.

Four moveable indices (bugs) of different colors are located on the bezel glass ring for placement to critical airspeeds.

Altimeters

Pilot Altimeter

The pilot altimeter is actually an electrical altitude indicator driven by signals from the air data computer, which senses the barometric data and provides it to the altimeter in electronic outputs. The air data computer is pre-programmed with altimeter installation error corrections, so it is not necessary to apply altimeter corrections at any altitude or airspeed. AC power is required for operation of the altimeter.

The altitude alert annunciator light illuminates to provide a visual indication when the aircraft is within 1,000 feet of the preselected altitude and extinguishes when the aircraft is within 250 feet of the preselected altitude. The light illuminates if the aircraft departs more than 250 feet from the selected altitude.

Copilot Altimeter

The copilot altimeter is a conventional barometric type with a counter/pointer readout. An optional encoding altimeter, which pro-



vides an alternate source of uncorrected barometric altitude information for transponders, is available for the copilot. In installations having the optional copilot encoding altimeter, a switch is provided on the instrument panel to control selection of the source of altitude information which is transmitted to the transponders. The copilot altimeter has a dedicated static system which operates from the forward left and forward right static ports. If SB650-34-107 is incorporated, the copilot altimeter will be supplied information by the copilot air data computer. In this case, a red OFF flag will be displayed in the event a power failure occurs.

Ram-Air Temperature (RAT) Indicator

The ram-air temperature (RAT) indicator is located above the copilot airspeed indicator (Figure 16-65). It indicates ram air (uncorrected) temperature. The indicated temperature must be corrected for compressibility effect to arrive at ambient or outside air temperature (OAT).

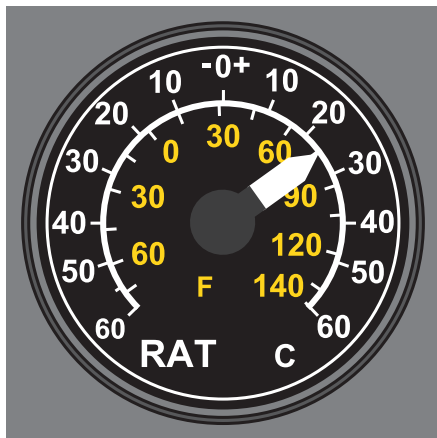


Figure 16-65. Ram Air Temperature Indicator

TAS/SAT/TAT Indicator (Optional)

The digital true airspeed/temperature indicator is optional and may be oriented horizontally or vertically. It may be located on the pilot subpanel or center instrument panel (Figure

16-66). The true airspeed (TAS) indicator displays true airspeed in knots and static air temperature (SAT) or total air temperature (TAT) in degrees Celsius. Static air temperature (SAT) is the temperature of air undisturbed by the motion of the aircraft. As SAT is read from the instrument, it has been corrected for the ram-air temperature rise and for compressibility effects. It is analogous to ambient air temperature or outside air temperature (OAT). Total air temperature (TAT) is ambient air temperature which has been compressed and has consequently has its temperature increased by adiabatic heating. When TAT is selected, therefore, the indicator registers the temperature of the air which results from the effect of ram air on the temperature probe. The indicator normally displays true airspeed and static air temperature. Pressing the TAT button changes the temperature display to total air temperature for as long as the button is depressed.

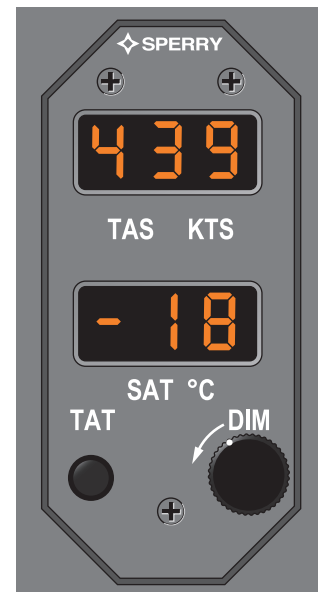
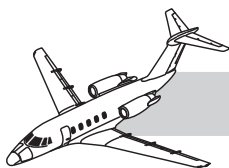


Figure 16-66. Optional TAS/SAT/TAT Indicator

VN-800 VNAV Computer/Controller

The VNAV computer/controller (VNCC) provides the data inputs for the altitude preselect mode (ALTSEL), altitude alert, and vertical navigation (VNAV) mode. Data is entered into



the computer by rotating the data select switch to the desired position and then setting the required value with the data set knob. The profile mode is selected and annunciated by the green illuminated profile select switch. It is a push-on/push-off switch which is illuminated when VNAV profile mode is selected. The VNAV operates only when using NAV 1 with the DME set to NAV 1 and locked on the selected VORTAC (not in HOLD). VNAV information is valid only when flying directly TO or FROM a VORTAC.

Vertical Speed Indicators

Two instantaneous vertical speed indicators indicate vertical velocity from 0 to 6,000 feet per minute, either up or down. Their operation differs from conventional vertical speed indicators in that there is zero time lag between aircraft displacement and instrument indication. Accelerometers sense any change in normal acceleration and displace the needle before an actual pressure change occurs. The instruments are pneumatic.

EDZ-600/800 SERIES ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS) (OPTIONAL)

General

If installed, the electronic flight instrument system consists of two electronic displays (EDs), one symbol generator (SG), one display controller (DC), and one source controller (SC) per pilot position (Figure 16-67).

The electronic displays (Figure 16-68) are identical and interchangeable (EADIs/EHSIs). A conventional slip/skid indicator is attached to the top cathode-ray tube (EADI). Both electronic displays use a combination of manual and photoelectric dimming for various light conditions.

The symbol generator is the heart of the system and receives all the aircraft sensor inputs. The



Figure 16-68. Pilot EFIS Displays

sensor information is processed and transmitted to the electronic displays.

The display controller (Figure 16-69) provides the means by which the pilot can control the display formatting, such as full or partial compass display or single-cue or crosspointer display. Also included on the display controller are the heading and course select knobs.



Figure 16-69. Optional EFIS Display Controller

The source controller (Figure 16-70) is used to select attitude, heading, navigation, and bearing sources for display. Annunciations of the selected sources are presented on the electronic displays.

Operation of the EFIS is similar to the standard flight director system except for the presentation of additional information on the two electronic display units (EADI and EHSI). All


CITATION 650 SERIES PILOT TRAINING MANUAL


Figure 16-67. Flight Guidance System: Primarily SNs 0001—0178 (Optional EFIS)

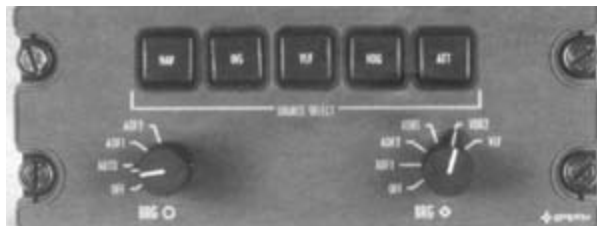


Figure 16-70. Optional EFIS Source Controller

information is not displayed at the same time. The presence or absence of each display element is determined by flight phase navigation radio tuning, selected flight director mode, absolute altitude, etc.

NOTE

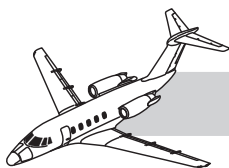
EDZ-601 system utilizes standard (4.6 x 5-inch) displays.

EDZ-801 system utilizes optional (5 x 6-inch) displays.

Dual EFIS systems are the most common configuration, in which case the copilot will have displays and controls identical to those of the pilot.

Composite Format

In the event of an EADI/EHSI display failure, the REV button on the display controller (Figure 16-69) may be used to display a composite



format on the remaining good display. The first push of the button blanks the EHSI and puts a composite display on the EADI. The second push of the button blanks the EADI and puts a composite display on the EHSI. The third push of the button returns the display to normal operation.

Reversionary Modes

In the event of an SG-601 symbol generator failure, pressing the REV button three times enables cross-side display information to be transferred to the on-side EADI/EHSI display. The reversionary mode is annunciated on the pilot and copilot EADIs as SG1 or SG2, depending on whether the source is the No. 1 (pilot) or No. 2 (copilot) symbol generator. Pushing the REV button a fourth time reverts the EADI and EHSI displays back to normal operation.

The HDG and ATT buttons on the source controller (Figure 16-70) are utilized to select an alternate source of heading and/or attitude information. In the event of a compass failure (HDG flag) on the EHSI, push the HDG button to select the opposite side C-14D compass to drive the EHSI. The heading reversionary mode is annunciated on the EHSIs as amber CMP1 or CMP2, as appropriate. Failure of a vertical gyro is annunciated on the EADI with an ATT failure flag, and the flight director command cue and gyro display disappear. Push the ATT button; the opposite-side VG now drives both EADIs, the gyro sphere and command cue reappear, and VG 1 or 2 is annunciated on the EADIs in amber.

Multi-Function Display (MFD) System (Optional)

An optional MDZ-816 Multi-Function Display system (Figure 16-71) is installed on the lower center instrument panel. The MFD display serves as a radar indicator or as a backup to the EFIS system. The MFD symbol generator can be used to back up the EFIS symbol generators. The MFD display tube can also be used to back up the EHSI display tubes. The MFD system expands on the navigation mapping capability of the EFIS systems.



Figure 16-71. Multi-Function Display

The MFD system is controlled by an MFD controller installed on the center pedestal. The controller is used to select various modes of operation: MAP, PLAN, weather, checklist (normal and emergency), and EFIS backup modes.

C-14D Compass Systems on EFIS-Equipped Aircraft

The pilot EHSI, the copilot RMI, and the pilot flight director are driven by the pilot No. 1 C-14D slaved gyro system. The system consists of a directional gyro, a flux detector, a mode selector switch, a remote compensator, and a slaving indicator on the EHSI. The pilot system is powered by the emergency DC electrical system.

The LH GYRO SLAVE switch, located on the lower left switch panel (Figure 16-72), has two position, labeled MAN and AUTO, and allows the compass to be operated in the slaved or free DG mode. In the AUTO (slaved) mode, the compasses align at approximately 3 to 5° per second. When MAN is selected, the EHSI and the copilot compass card can be moved left or right at a rate of 30° per minutes by toggling the LH–RH switch. In this mode, the slaving indicator on the EHSI disappears. Under normal operating conditions, the gyros remain in the AUTO (slaved) mode.

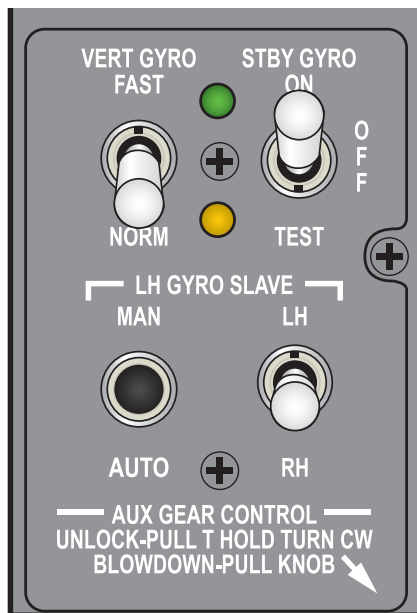


Figure 16-72. Pilot Lower Instrument Switch Panel

The copilot C-14D compass system is identical to the pilot system. The copilot system drives the right-side EHSDI, the pilot RMI, and the copilot flight director. The copilot C-14D compass system is powered from the main DC bus. (Chapter 2, “Electrical Power Systems,” of this manual for emergency operating procedures.)

Emergency Power Conditions

Operating on emergency DC electrical power only, the pilot RMI is operational (driven by the No. 1 C-14D). During a loss of both inverters (normal DC power still available), the pilot RMI compass card and No. 1 needle are operational, provided the FD 1 circuit breaker (DC) on the copilot circuit breaker panel is pulled.

STANDBY FLIGHT INSTRUMENTS

Standby Attitude Gyro

A standby attitude indicator (Figure 16-73) is located on the pilot instrument panel. It normally operates on emergency DC electrical power through the STDBY ATT IND or STDBY GYRO circuit breaker on the right circuit breaker panel. Power to the gyro is

controlled by the STBY GYRO switch, with ON, OFF, and TEST positions, located on the pilot lower instrument panel (Figure 16-72). An emergency battery pack in the nose avionics compartment is an emergency source of power for the standby gyro if main DC bus voltage falls below minimum. This is indicated by an amber power-on light adjacent to the STBY GYRO switch, provided the switch is in the ON position. The battery pack also provides power for emergency instrument lighting for the pilot primary flight instruments.

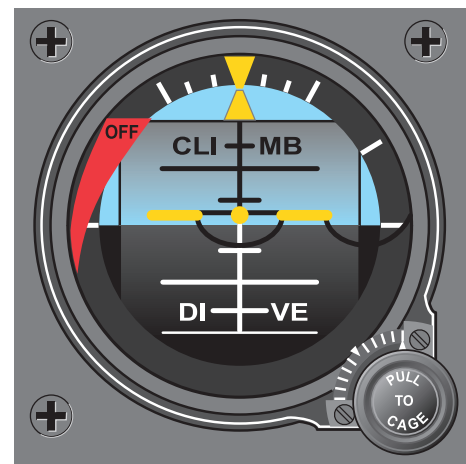
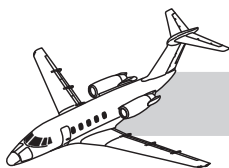


Figure 16-73. Standby Attitude Indicator

The battery pack is continuously charged by the emergency DC electrical system and should be fully charged in case of electrical power failure. The STBY GYRO power switch must be in the ON position for automatic transfer to emergency battery pack power. The gyro will operate for a minimum of 30 minutes on emergency battery pack power. The amber light adjacent to the switch illuminates when the gyro is operating on its own power source (battery pack). When the switch is held to the TEST position, a self-test of the emergency battery pack and associated electrical circuits is accomplished. The amber light adjacent to the switch illuminates if the test is satisfactory and the battery pack is fully charged. The green light illuminates with the switch ON and aircraft power operating the standby gyro and charging the emergency battery pack.

The standby gyro is caged by pulling the PULL TO CAGE knob and rotating it clockwise.



CAUTION

When uncaging, do not release the PULL TO CAGE knob suddenly so that it snaps back; this may damage the gyro.

COMMUNICATION/ NAVIGATION EQUIPMENT

VHF COMM TRANSCEIVERS

Dual VHF-22A transceivers are located in the nose avionics bay. They are individually con-

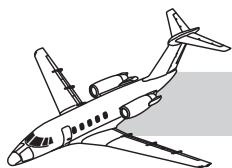
trolled by CTL-22 control heads located on the right side of the center instrument panel (Figure 16-74). Each unit is a 720-channel, VHF receiver-transmitter with a frequency range from 118.000 to 135.975 MHz. The COMM 1 radio is power from the emergency DC bus.

VHF NAVIGATION RECEIVERS

Dual VIR-32 navigation receivers provide VOR, localizer, glideslope, and marker beacon capability. The receivers are located in the nose avionics compartment. CTL-32 control heads are located on the lower right side of the center instrument panel (Figure 16-74). Each system has 200 VOR/LOC operating channels and 40 glideslope channels and automatic DME channeling. Multiple outputs drive the flight direc-



Figure 16-74. Communications/Navigation Control Panel



tor, EHSIs, RMIs, autopilot, and long-range navigation system (FMS). All basic functions have a built-in self-test. Consult Section III of the *Airplane Operating Manual* for self-test procedures. The NAV 1 receiver is powered from the emergency DC bus.

AUTOMATIC DIRECTION FINDER (ADF)

A Collins ADF-462 is installed in the nose avionics bay and is controlled by a CTL-62 electronic control head mounted on the right side of the center instrument panel (Figure 16-74). The control head has two digital readouts to display the active frequency and a preset frequency. Four additional frequencies may be stored in memory. ADF magnetic bearings are displayed on the RMIs and on the EHSIs. An optional additional ADF may be installed, in which case the operation is identical to that of the No. 1 system. The No. 1 ADF bearings are displayed on the RMIs by the single-bar bearing pointers and on the EHSIs by the single-bar (blue) bearing pointers. The No. 2 ADF (if installed) bearings are displayed by the double-bar bearing pointers on the RMIs and the double-bar bearing pointers (green) on the EHSIs.

RADIO MAGNETIC INDICATORS (RMIs)

Dual RMI-36 radio magnetic indicators are mounted on both instrument panels. ADF and VOR magnetic bearing information is displayed on each RMI. Pushbutton selector switches for each pointer are located on the lower case of the RMIs. The compass card of each RMI is normally driven by the opposite-side compass system. (See discussion of RMIs under Flight Guidance Systems, this chapter, for further information.)

HF COMMUNICATION (OPTIONAL)

An optional KHF-950, 150-watt transceiver that provides 280,000 frequencies at 100-Hz increments with 99-channel preset capabilities

in the HF band (2 to 29 MHz) may be installed. It operates in AM or single sideband. A KFS-954 compact control panel is installed, normally on the copilot instrument panel, to operate the HF communications system (Figure 16-75). An optional second KHF-950 may be installed, in which case the KFS-954 control panel will normally be mounted on the pilot lower instrument panel.

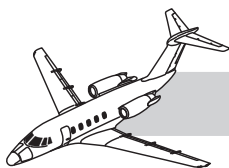


Figure 16-75. Optional HF Communications Radio

An HF transfer option (HF XFER) may be installed. The HF XFER pushbutton annunciator is provided on the panel next to the HF control. The HF XFER option provides the capability of transferring HF communications to the cabin interphone system. Upon pressing the HF XFER button, an indicator light on the switch illuminates and remains on until the passenger hangs up his telephone. This option allows a passenger to use the marine radio-telephone system.

NOTE

Some individual aircraft may have different designated equipment. If so, they all operate essentially the same. Consult the *Operating Manual* for detailed operating instructions.



EMERGENCY LOCATOR TRANSMITTER (ELT) (OPTIONAL)

The optional locator beacon system consists of an emergency locator transmitter (ELT) designed to assist in locating a downed aircraft. The ELT has a self-contained battery pack which must be replaced every three years or when one cumulative hour of operation is logged.

The ELT is activated automatically by a $5 +2/-0$ g impact along the flight axis of the aircraft, or manually by the ELT BCN (EMERG-NORMAL) switch located behind the copilot circuit breaker panel (Figure 16-76). The EMERG position activates the ELT, and the NORMAL position arms the impact switch. The switch is in NORMAL with the guard closed. The BCN RESET button is used to reset the ELT if it has been energized by the impact switch. BCN RESET will turn off the transmitter and rearm the impact switch. The BCN RESET button should be held for a minimum of three seconds to reset the ELT.



Figure 16-76. Optional ELT Beacon Control

A remote control, accessible from outside the aircraft, is located on the right aft fuselage under a plug button. Pull out the plug button; then the ELT can be activated, turned OFF, or RESET from the remote control.

The ELT is installed in the tailcone on the right side of the fuselage. It can be removed by turning a wingnut, removing the ELT mounting bracket, and disconnecting the antenna and wiring connectors. It can be used as a portable installation since its battery pack is self-contained and a master switch is included on the transmitter. Two aircraft antennas for the ELT are mounted on top of the aft fuselage just forward of the vertical fin, one left and one right.

The ELT, when activated, transmits a modulated omnidirectional signal simultaneously on emergency frequencies 121.50 and 243.00 MHz. The modulated signal is a downward swept tone which starts at 1600–1200 Hz and sweeps down to 700 Hz every two to four seconds continuously and automatically.

AUDIO PANEL CONTROLS

Two audio control panels (Figure 16-77) provide individual audio selection by each pilot. Three-position switches labeled SPKR, OFF, and HDPH enable all audio inputs to be selected to the overhead speakers or headphones. A two-position IDENT-VOICE switch is used with the NAV and ADF switches to monitor either voice or coded identifiers. The MASTER VOLUME knob controls the headset or speaker volume of all selected audio sources. A PASS SPKR VOLUME knob controls the output volume of the passenger compartment speakers.

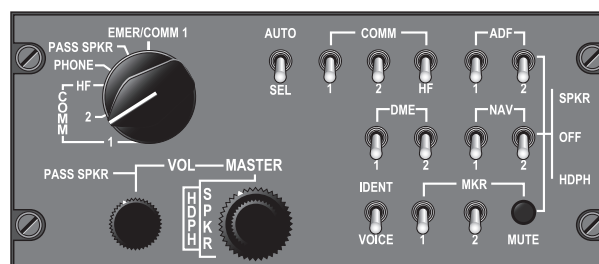
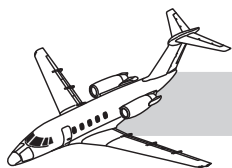


Figure 16-77. Audio Control Panel

A rotary microphone selector switch has four standard positions: COMM 1, COMM 2, PASS SPKR, and EMER/COMM 1. A fifth position, labeled HF, is included if an optional HF radio



CITATION 650 SERIES PILOT TRAINING MANUAL

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is installed. COMM 1 or COMM 2 connects the microphone being used to the respective VHF transmitter. PASS SPKR provides for announcements to the passengers through the cabin speakers; COMM 1, COMM 2, and HF audio is muted.

On SNs 0001—0178, an EMER/COMM 1 position provides for the use of COMM 1 when operating only on emergency DC power. The EMER/COMM 1 position bypasses the audio amplifier, necessitating the use of a headset, and volume control is available only at the radio control head. Transmitting remains normal from all microphone sources.

A three-position AUTO SEL switch with SPKR, OFF, and HDPH positions automatically selects the proper speaker or headphone to match the position of the rotary microphone selector switch. All audible sources can be monitored at any time by the use of the appropriate SPKR—OFF—HDPH switch, regardless of the microphone selector switch or the AUTO SEL switch positions. A MKR MUTE button silences the marker beacon audio for approximately 30 seconds.

If the cabin interphone is installed, the cabin is provided with a handset, which, if lifted, sounds a musical tone to alert the crew that communication is desired. A cabin call is provided on the audio panels to signal the passenger compartment that communication is desired.

A two-position switch, one on each control wheel, has a MIC position for keying the transmitters and an INPH position for interphone communications when using the lip phone or the oxygen mask microphone. If a hand-held microphone is used, transmission is determined by the position of the MIC selector switch.

SNs 0179 and Subsequent

The pilot audio panel is powered from the emergency DC bus. The copilot audio panel is powered from the right DC extension bus. However, if main DC power is lost, an emergency audio relay closes to switch the No. 2 AUDIO circuit breaker to the emergency DC

bus. Circuit breakers labeled AUDIO 1 and AUDIO 2 are located on the right circuit breaker panel. In the event main DC power is lost, the audio panels and the overhead speakers continue to operate (COMM 1 and NAV 1 audio will be received).

Honeywell Primus II Remote Radio System (Optional)

The Honeywell Primus II remote radio system may be installed as an option. If this system is installed, the VHF communication, navigation, ADF, transponder, and DME control heads are replaced with programmable CRT tubes. The standard audio control panel is replaced by Primus II audio control panels. Consult Section III of the *Airplane Operating Manual* and the Honeywell pilot handbook for operating instructions.

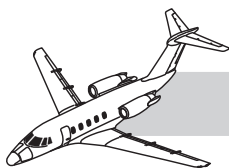
Flitefone VI

The Flitefone system provides air-to-ground telephone communication. It operates in the UHF band and is a frequency modulated (FM) unit. Operating frequency is in the 450-MHz range. Twelve telephone channels are provided plus one ground-to-air signaling channel.

The Flitefone system may be mounted in the cockpit on the center pedestal or to the right of the copilot on the cockpit/cabin divider panel.

The standard passenger compartment Flitefone consists of a cabin control and bell assembly (Figure 16-78). The handset is equipped with a push-to-talk (PTT) button for use with the HF radio (if HF is installed). The HF mode of operation is selected with a slide switch on the control which removes the handset connections from the Flitefone system and connects it to the HF transceiver.

The Flitefone also serves as a cabin/cockpit interphone. The interphone system becomes operational on controls with a dedicated INTERCOM (IC) button when either IC button is pressed or 4 (I)-2(c)-#(enter) key is dialed.

**Figure 16-78. Flitefone VI**

For controls without a dedicated intercom button, the 4 (I)-2(c)-# key must be dialed. The amber intercom indicator (LED) is illuminated when an IC key is depressed or when the system is in intercom. The indicator is also illuminated when all system control units are on hook and a call is not being processed. The intercom system can be used at any time.

To activate the voice privacy function of the Flitefone VI, place a direct-dial call and press 8(V)-7(P)-# key. A slight warbling in the background will be heard, or if voice privacy is unavailable, five rapid “beeps” will be heard. To switch off voice privacy, press 8(V)-7(P)-# key, and the voice privacy function disengages after ten seconds. Voice privacy automatically disengages at the end of a call if the operator hangs up the handset.

For more extensive information concerning programming and operation of Flitefone VI and IV or V, consult Section III of the *Airplane Operating Manual* and the latest Global Wulfsberg Systems Flitefone operator’s manual.

PULSE EQUIPMENT

Transponders

The transponder installation is two Collins TDR-94 transponders, each with 4096 Mode A and Mode C code capability. They are controlled by a single CTL-92 electronic controller mounted on the center instrument panel (Figure 16-74). Altitude information may be obtained from either the pilot or copilot digital air data

computer, by a selector switchlight located next to the transponder controller. The transponders also have Mode S capability (collision avoidance information to ground station radar). Any time the code select knob is at ON or ALT, Mode S will be transmitted if the ground station also has Mode S capability.

The digital controller has a means of selecting new codes and for storing a preset code. To insert a code in memory, push and hold the PRE button while turning the code select knobs. To recall the stored code, momentarily press the PRE button again.

A two-position 1/2 switch on the controller selects the No. 1 or No. 2 transponder. An IDENT button on the control head and on each control wheel is used to “squawk ident” when requested, and the letters “id” appear in the lower window for approximately 15 seconds. An RMT annunciation indicates that the transponder is being remotely tuned. TX illuminates each time the transponder replies to an interrogation.

A self-test is initiated by selecting the TEST position with the mode switch ON and a code selected. ALT is displayed in the upper window, and the altitude in thousands of feet is displayed in the lower window (Figure 16-74).

Transponder equipment installed in Citation aircraft may differ slightly. Refer to the *Airplane Operating Manual* for detailed operating instructions for transponders.

Distance Measuring Equipment (DME)

DME-42

The DME installation consists of one DME-42 receiver-transmitter and one IND-42A indicator (Figure 16-79). Dual DME-42s and dual IND-42As may be installed as an option. The IND-42A indicator does not control selection of DME data. It is used only to display data that has been selected by the NAV receivers. Depressing the CH button alternately selects between NAV 1 and NAV 2. DME information



is displayed on the EHSIs. Depressing the NAV SEL button on the EFIS display controller or source controller, or the flight director mode selector for standard electrical/mechanical systems, determines which NAV receiver is providing the distance readout. The mode selector (SEL) switch sequentially selects KT (knots), MIN (minutes-to-station), and ID.



Figure 16-79. DME-42 Control Unit

DME-40

Earlier model Citation III aircraft may have the DME-40 receiver-transmitter system (Figure 16-80). This system operates essentially the same as the DME-42 system. Consult the *Airplane Operating Manual* for detailed operating instructions.



Figure 16-80. DME-40 Control Unit

Radio Altimeter

Radio altitude is displayed in the lower right corner of the pilot and copilot ADIs and EADIs. The altitude displays from -20 to 2,500 feet. Between 200 and 2,500 feet, the display is in ten-foot increments. Below 200 feet, it is in 5-foot increments. Above 2,500 feet the display disappears. A rising runway appears on the EADI when absolute altitude of less than 200 feet is reached.

Decision height is set by the inner concentric knob labeled "TEST" on the EFIS display controller. Decision height is displayed in the window on the lower left corner of the EADIs. The decision height aural warning is controlled by the DH setting on the pilot EADI. The copilot setting has no effect on the aural warning. As the aircraft descends below an altitude of 100 feet above the selected decision height, a white box appears on the EADI, and when the decision height is reached, an amber DH appears inside the box.

On conventional ADIs, the decision height is set by a knob on the lower right side of the instrument and displayed on the lower left window. Radio altitude is displayed (below 2,500 feet) in the lower right window.

Conventional Radio Altimeter (Optional)

If an optional conventional radio altimeter indicator(s) is installed, it is operated by the same radio altimeter, but the conventional display operates independently (Figure 16-81).

Weather Radar— Primus 870 Color Radar

The Primus 870 digital weather radar system is an advanced multicolor radar with traditional weather displays plus the addition of turbulence detection. The system detects targets of varying rainfall intensity, as well as the random motions of raindrops caused by turbulent air. The standard radar installation is mounted on the center instrument panel and includes the radar indicator and indicator-mounted con-

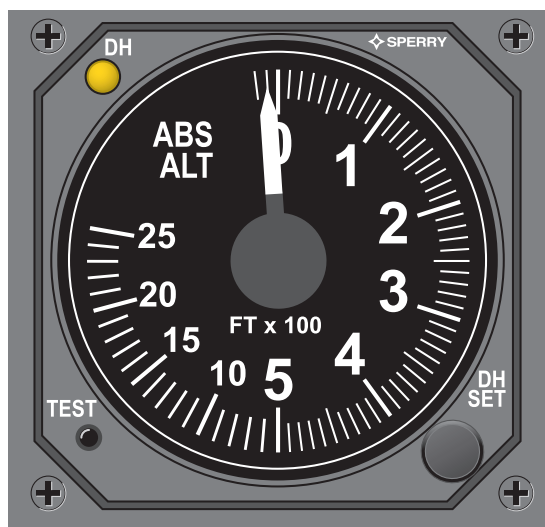
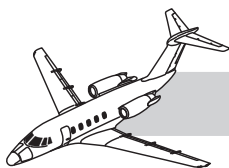


Figure 16-81. Conventional Radio Altimeter Indicator

trols. On aircraft equipped with the optional multifunction display (MFD), the MFD replaces the radar indicator, and the WC-870 radar controller is located on the center pedestal (Figure 16-82). Dual WC-870 controllers are also available.

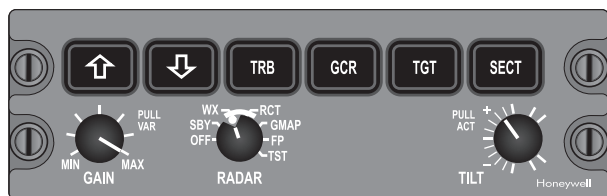


Figure 16-82. WC-870 Color Radar Controller

The system can be operated in conjunction with the EFIS and the MFD equipment to provide radar video displays. Storm intensity is displayed at five color levels, with black representing weak or no returns and green, yellow, red, and magenta showing progressively stronger returns. Areas of high turbulence are displayed in soft gray to white.

WARNING

The 870 radar system cannot detect clear air turbulence.

In ground-mapping mode, video levels of increasing reflectivity are displayed as black, cyan (sky blue), yellow, and magenta. The ground-mapping mode (GMAP) permits display of prominent landmarks such as lakes, bays, islands, shorelines, high ground, large populated cities, etc.

WARNING

The system displays only weather and ground-mapping returns. It is not to be used or relied upon for proximity warning, anticollision, or terrain avoidance.

NOTE

Antenna scan and transmitter are inhibited on the ground. However, simultaneously pressing both range buttons will allow normal operation on the ground.

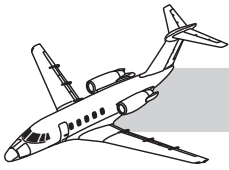
WARNING

Selecting TST on the controller selects the radar test mode. The transmitter is on and radiating in TEST. The receiver-transmitter antenna is located in the forward nose section.

Other radar installations may be installed, especially on earlier 650 model aircraft. Consult specific vendor handbooks and the *Airplane Operating Manual* for detailed operating instructions.

AREA NAVIGATION—FLIGHT MANAGEMENT SYSTEM (FMS)

Various FMS systems are installed in the aircraft. For more extensive information concerning programming and operation of installed flight management systems consult Section III of the *Operating Manual* and specific vendor handbooks.



MISCELLANEOUS EQUIPMENT AND INSTRUMENTS

TURN AND BANK INDICATIONS

Both EADIs are provided with a built-in turn needle which is displayed at all times except when an ILS frequency is tuned in on a NAV radio and selected on the EFIS display controller. Conventional inclinometers are attached to the lower edge of the EADI case. ADIs associated with standard electrical/mechanical flight directors (SNs 0001—0178) have separate turn needles and localizer course indicators.

MAGNETIC COMPASS

A standard liquid-filled magnetic compass is mounted above the glareshield (Figure 16-83).



Figure 16-83. Magnetic Compass

FLIGHT HOURMETER

The flight hourmeter, located just forward or aft of the copilot circuit breaker panel, displays total aircraft flight time in hours and tenths (Figure 16-84). Either landing squat switch activates the meter when the weight is off the gear. A small indicator on the face of the instrument rotates when the hourmeter is in operation.



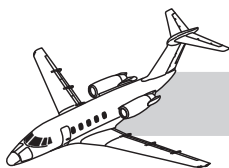
Figure 16-84. Flight Hourmeter

DIGITAL CLOCK

The Davtron model M877 clocks (Figure 16-85) can display four functions: local time, GMT, flight time, and elapsed time. Two versions of the elapsed time function may be selected; count up or count down. The clocks are mounted on the pilot and copilot instrument panels.



Figure 16-85. Davtron Clock



PITOT-STATIC SYSTEM

GENERAL

The pitot-static system supplies dynamic and static air pressure for operation of the air data computers, Mach/airspeed indicators, altimeters, vertical speed indicators, Mach/airspeed warning switch, cabin differential pressure indicator, and the pressurization computer/controller. SNs 0001—0178, 0200—0202, and 0207—0241 incorporating SB650-34-105 or SB650-34-107 will have an air data computer installed/integrated into the copilot side pitot-static system. Consult applicable service bulletin and AFM supplement for additional information.

PITOT TUBES

The pitot tubes are mounted on each lower side of the fuselage nose. They provide independent supplies, as shown in Figure 16-86.

Both pitot tubes are electrically heated. Pitot heat is controlled by two PITOT/STATIC LH and RH switches located in the ANTI-ICE group on the center instrument panel (Chapter 10, “Ice and Rain Protection,” for additional information).

The left and right pitot tubes provide pitot pressure to their respective digital air data computer (ADC). The right pitot tube also provides pressure to the standby airspeed indicator on the pilot instrument panel in SNs 0179—0199, 0203—0206, and 7001—7119.

STATIC PORTS

Upper, lower, and forward static ports are located on each side of the fuselage adjacent to the aft cockpit side windows. As shown in Figure 16-86, dual pickups are provided to the pilot and copilot instruments from both sides of the aircraft. The dual pickups are provided to reduce sideslip effects on the airspeed indicators, altimeter, and vertical speed indicators. All static ports are heated and controlled by the PITOT/STATIC LH and RH

ANTI-ICE switches. The third (forward) set of ports provides static pressure to only the standby pneumatic function of the pilot altimeter or copilot altimeter.

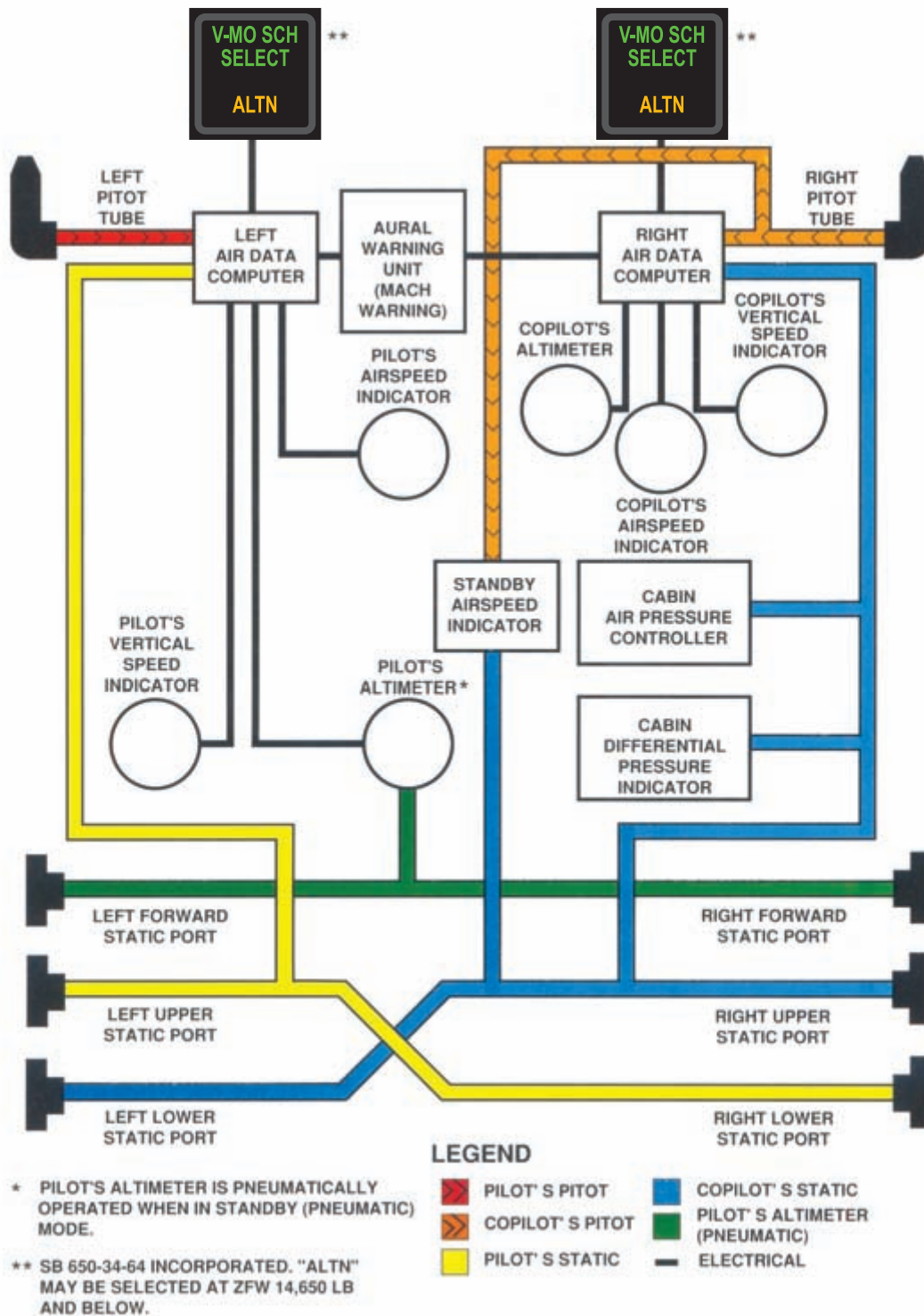
STATIC DISCHARGE WICKS

A static electrical charge builds up on the surface of an aircraft while in flight and causes interference in radio and avionics equipment operation. The charge may be dangerous to persons disembarking after landing, as well as to persons performing maintenance on the aircraft. The static wicks are installed on all trailing edges and dissipate the static electricity in flight.

There are a total of 23 static discharge wicks installed on the aircraft. No more than a total of two static discharge wicks may be missing from the airframe to dispatch for flight. There cannot be two adjacent static discharge wicks missing at any one time.

COCKPIT VOICE RECORDER

A GA-100 cockpit voice recorder provides a continuous 30-minute record of all voice communications from the cockpit, as well as sounds from warning horns and bells. A sensitive microphone is located in the copilot instrument panel to the right side of the annunciator panel. The system is energized by main DC power. The control panel, (Figure 16-87) located on the center pedestal forward of the throttle quadrant, contains a TEST button and an ERASE button. Depressing TEST for five seconds illuminates the green light on the control panel to indicate proper functioning of the recorder. Pressing ERASE for approximately two seconds causes the entire record to be erased. Erasure can be accomplished only on the ground with the main cabin door open.

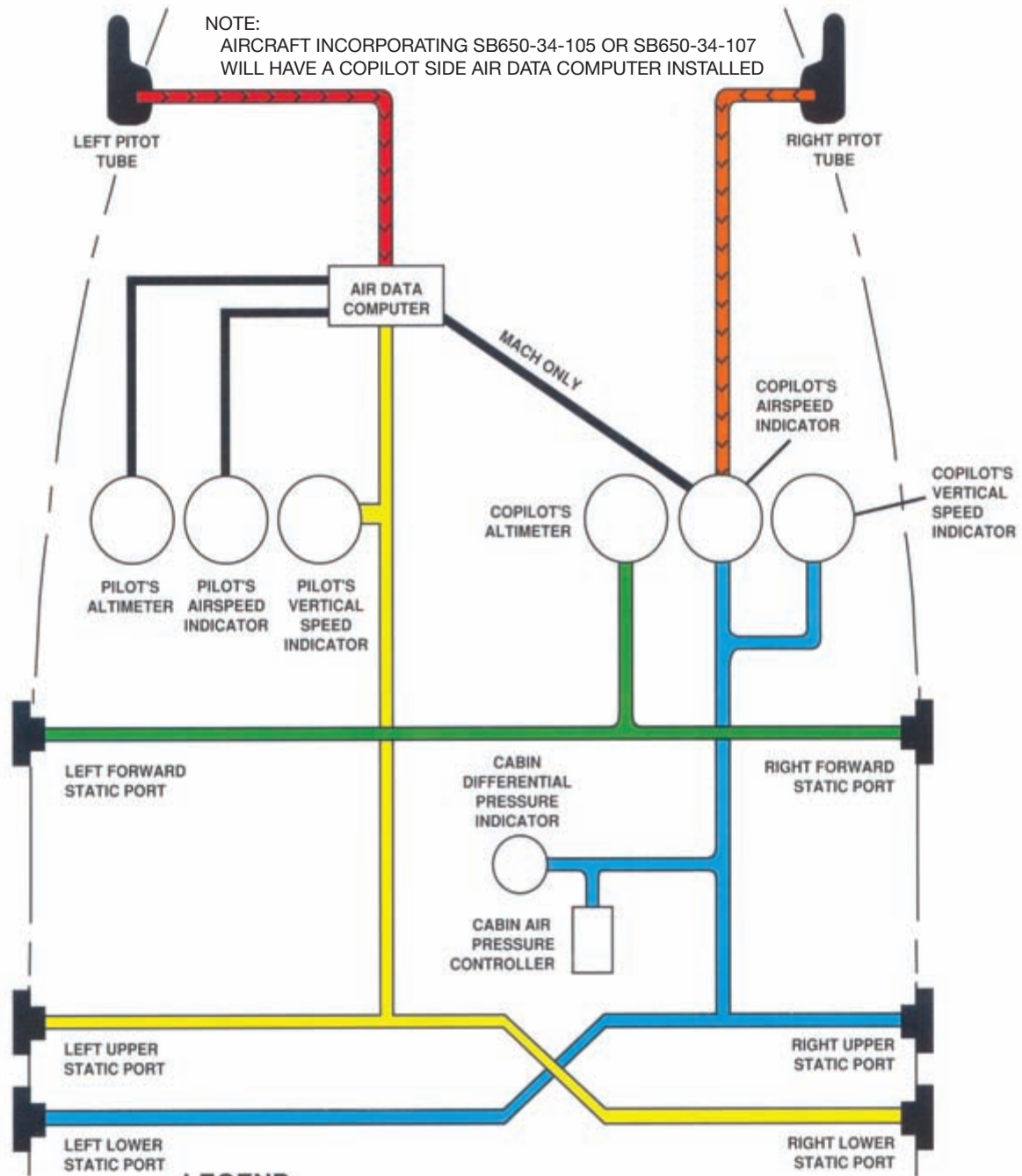

CITATION 650 SERIES PILOT TRAINING MANUAL


SNs 0179—0199, 0203—0206, and 7001—7119

Figure 16-86. Pitot-Static System (1 of 2)



CITATION 650 SERIES PILOT TRAINING MANUAL



LEGEND

- | | | | |
|--|-----------------|--|-----------------------------|
| | PILOT'S PITOT | | COPILOT'S STATIC |
| | COPILOT'S PITOT | | COPILOT'S ALTITUDE (STATIC) |
| | PILOT'S STATIC | | ELECTRICAL |

SNs 0001—0178, 0200—0202, and 0207—0241

Figure 16-86. Pitot-Static System (2 of 2)



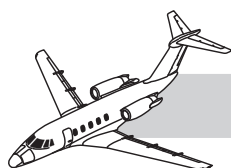
DIGITAL FLIGHT DATA RECORDER (OPTIONAL)

On aircraft built after October 11, 1991, that are configured with more than nine passenger seats and are operated under FAR Part 91 or FAR Part 135, a digital flight data recorder is required equipment. The flight data recorder continuously records at least 17 parameters of aircraft and systems operation. A continuous

recording of eight hours is required. Recorder operation begins with aircraft power-up and continues until electrical power is shut down. Recorder operation requires no attention from crew members. An annunciator (RECORDER PWR FAIL) on the instrument panel illuminates to advise the crew of recorder malfunction. The system is normally powered through an FDR circuit breaker on the right cockpit circuit breaker panel.



Figure 16-87. Cockpit Voice Recorder Control Panel



CHAPTER 17

MISCELLANEOUS SYSTEMS

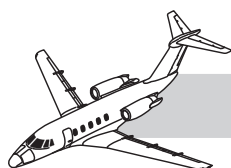
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CHAPTER 17

MISCELLANEOUS SYSTEMS



INTRODUCTION

The Citation 650 series aircraft provides various other systems for flight support and crew and passenger comfort, including an emergency oxygen system, various access doors, and a toilet. This chapter provides an overview of these features as well as information on general servicing of the aircraft.

OXYGEN SYSTEM

The oxygen system provides emergency oxygen for the cockpit and the passenger compartment. The oxygen system is a high pressure gaseous system. Crew oxygen is provided through standard oxygen masks or optional EROS oxygen masks, both of which are clas-

sified as quick-donning. Passenger oxygen is provided through overhead dropdown oxygen masks, activated automatically when cabin pressure drops or when the masks are selected by a valve in the cockpit. A standard walk-around oxygen bottle is also provided.



COMPONENTS

Storage Bottle

The oxygen system includes a 49 or 76 cubic foot bottle with the option to change from the smaller to larger size bottle. The bottle is stored in the lower left side of the nose compartment.

Either bottle is charged to 1,850 psi. The pressure is reduced to 70 psi through a supply regulator for crew and passenger distribution. The oxygen bottle is serviced through a filler connection inside the left nose compartment access door (Figure 17-1). A green disc is in the end of the bottle overpressure vent line and flush-mounted on the left side of the aircraft nose (Figure 17-1). If an overpressure condition in excess of 2,500 psi occurs, the relief valve opens, allowing pressure to rupture the disc and empty all oxygen overboard.

NOTE

The oxygen relief valve activates only in the most adverse circumstances. If the disc is ruptured, determine the cause of the overpressure condition prior to flight.



Figure 17-1. Filler Valve and Oxygen Overpressure Disc

Oxygen Pressure Gauge

An oxygen pressure gauge on the pilot instrument panel (Figure 17-2) indicates oxygen bottle pressure. The pressure gauge is connected to the oxygen bottle below the shutoff valve and indicates pressure even when the valve is closed.

NOTE

Ensure that oxygen is available prior to flight. Place the oxygen mask regulator to the 100% position and then breathe through the mask, or place the regulator to the EMER position, and then listen for the rush of oxygen leaving the mask.



Figure 17-2. Oxygen Pressure Gauge

The oxygen pressure gauge has a graduated scale between 0 and 2,000 psi. Below 400 psi, the scale is yellow, which indicates that



the bottle requires maintenance. The normal operating range is 1,600 to 1,800 psi, indicated by the green scale. The red line on the gauge is at 2,000 psi, which indicates maximum bottle pressure.

Crew System

The crew oxygen system (Figure 17-3) distributes oxygen through the standard masks on the pilot and copilot consoles or the optional EROS mask on the cabin divider.

The standard and optional masks are both classified as quick-donning. The primary differences are the harness style and stowage method of each type of mask. Both masks have microphones. The oxygen mask microphones are controlled by two MIC OXY MASK-MIC HEAD SET switches on each side of the cockpit (Figure 17-4).

The MIC OXY MASK position activates the oxygen mask microphone and speakers, which receive audio when the interphone switch on the control wheel is activated.

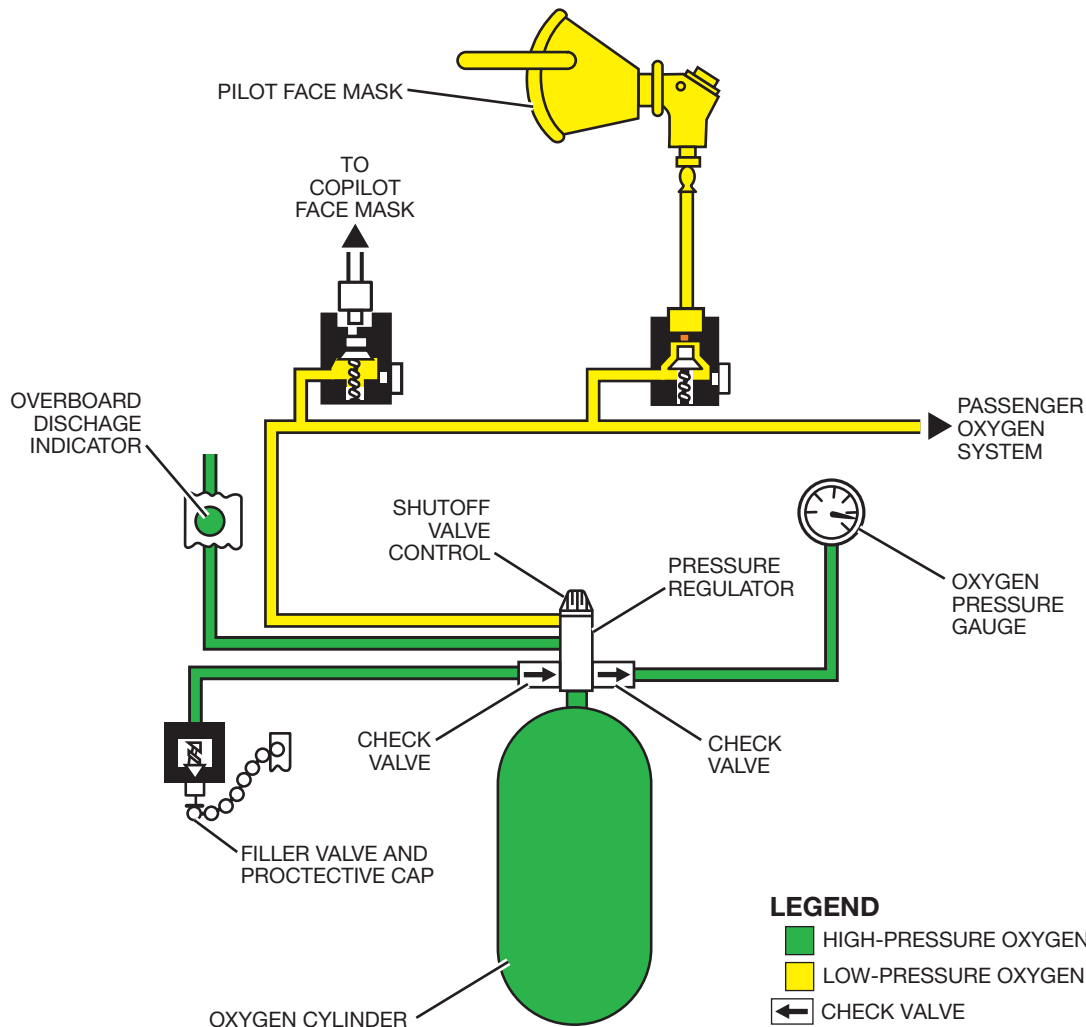


Figure 17-3. Crew Oxygen System



Figure 17-4. MIC OXY MASK Switch

NOTE

Wearing the oxygen mask interferes with wearing the headset. As such, the AUTO SEL switch(es) must be set to SPKR in order to hear audio inputs from the radio receivers.

Standard Masks

The standard mask (Figure 17-5) uses a bayonet locking connector, which fits into the side console and supplies 70 psi oxygen for the pressure demand mask. The crew mask bayonet fitting initiates oxygen flow when plugged into the side console. Removing the fitting terminates oxygen flow. The console connector is equipped with a dust cover to protect it when the mask is unplugged.

The integral oxygen regulator for the standard mask depicted in Figure 17-5 has a manual selector for NORM (diluter-demand), 100%, or EMER (pressure breathing) flow. The pilot is assured that oxygen is flowing when the regulator control is placed to EMER and a rush of air under pressure comes out of the mask.

With the regulator in the EMER position, the pilot is always furnished 100% oxygen under positive pressure. This protects the respiratory



Figure 17-5. Oxygen Mask Installation and Regulator

tract against smoke or obnoxious fumes at any altitude. In addition, above 37,000 feet cabin altitude, pressurized oxygen is automatically scheduled regardless of the regulator setting.

With the mask oxygen regulator in the NORM (diluter-demand) position, an increasing amount of oxygen is mixed with cockpit air as the cabin altitude increases to a maximum of 100% oxygen above 30,000 feet cabin altitude. NORM may be selected below 20,000 feet cabin altitude to conserve oxygen, and 100% should be selected above 20,000 feet cabin altitude to ensure adequate oxygen.

Optional EROS Mask

The optional EROS mask has a pneumatic head harness and a regulator stowed in a quick-release cup (Figure 17-6). The quick-release cup is on the cabin divider above the pilot and copilot seats. The regulator has a two-position N-100% PUSH rocker selector switch and a PRESS TO TEST/(turn for) EMERGENCY knob.



Figure 17-6. EROS Oxygen Mask (Stowed Position)

Pressing the red inflation levers on the side of the regulator automatically inflates the harness with oxygen pressure and releases the mask from the cup (Figure 17-7). After donning the mask, release the inflation levers to deflate the harness around the head.



Figure 17-7. EROS Oxygen Mask (Inflated Position)

The N (normal) position on the oxygen regulator makes available a mixture of cabin air and oxygen to a maximum of 100% oxygen above 30,000 feet cabin altitude. At cabin altitude above 37,000 feet, the supply is at positive pressure.

Selecting 100% provides pure oxygen at all altitudes. Positioning the oxygen regulator knob to EMER (emergency) or pressing the knob to the PRESS TO TEST position supplies pure oxygen at a positive pressure.

Stow the crew masks with the regulators in the 100% position, which makes 100% oxygen available immediately if rapid decompression, smoke, or fumes occur.

Pressure Breathing

Pressure breathing is difficult until one becomes accustomed to it. Normally, the human body must work to inhale and relax to exhale. This process is reversed with pressure breathing. When relaxed, the lungs are filled with oxygen under pressure, and the body must work to exhale. Pressure breathing makes radio transmissions difficult. Also, the oxygen mask must be tightened to counter the oxygen pressure.

Passenger Oxygen System

The passenger oxygen system (Figure 17-8) is controlled by a rotary selector knob (PASS OXY) on the pilot lower instrument panel (Figure 17-9). The selector knob has three positions: OFF–AUTO–ON and normally is positioned to AUTO. The ON position opens a manual valve, allowing oxygen flow to the passenger masks. The OFF position manually prevents oxygen flow to the passenger masks.

The AUTO position opens a mechanical valve that allows oxygen pressure to a flow to a normally closed solenoid valve. When cabin altitude reaches $13,500 \pm 500$ feet, the valve opens by normal DC power via a barometric pressure switch in the left console. The open valve allows oxygen flow to the passenger oxygen manifolds. The pressure switch opens, which deenergizes and closes the solenoid valve as cabin altitude descends through 8,000 feet.

Supplying the cabin manifolds with oxygen at 70 psi opens the passenger mask container doors, and the masks deploy (Figure 17-10).



NOTE

The PASS OXY rotary selector valve must be in the ON or AUTO position in order for the doors to open and the masks to deploy.

To conserve oxygen if all passenger masks are not needed, each mask is equipped with a lanyard and pintle pin. Pulling the lanyard

removes the pintle pin to allow continuous oxygen flow to the mask. Place the mask on the face and breathe normally. Smoking is prohibited when using oxygen.

When oxygen flow to the cabin stops, oxygen pressure bleeds down and the door actuators retract. After the door actuators retract, reinsert the pintle pins in the valves, stow the masks, and latch the oxygen mask doors.



Figure 17-9. PASS OXY Knob

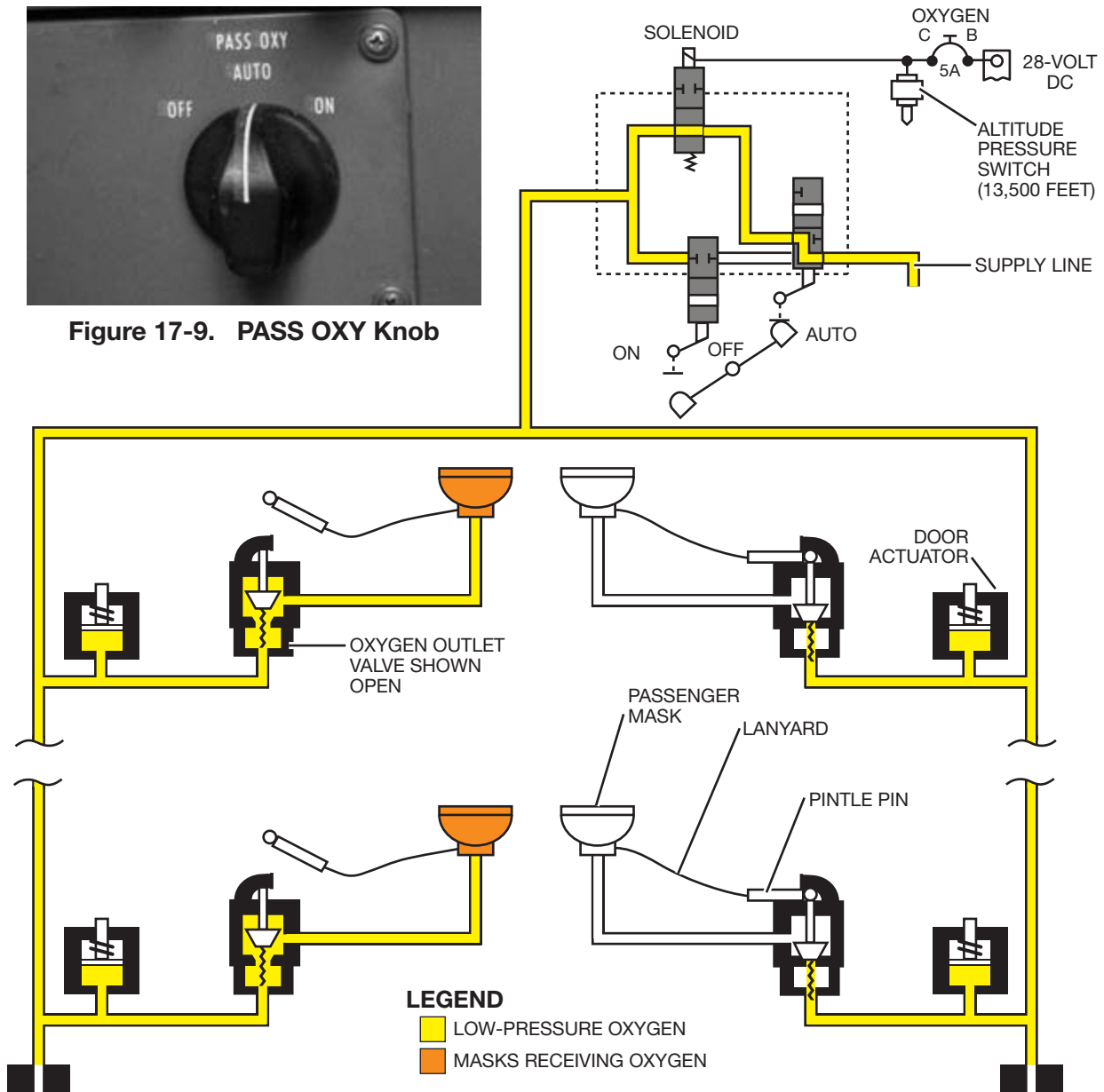


Figure 17-8. Passenger Oxygen System

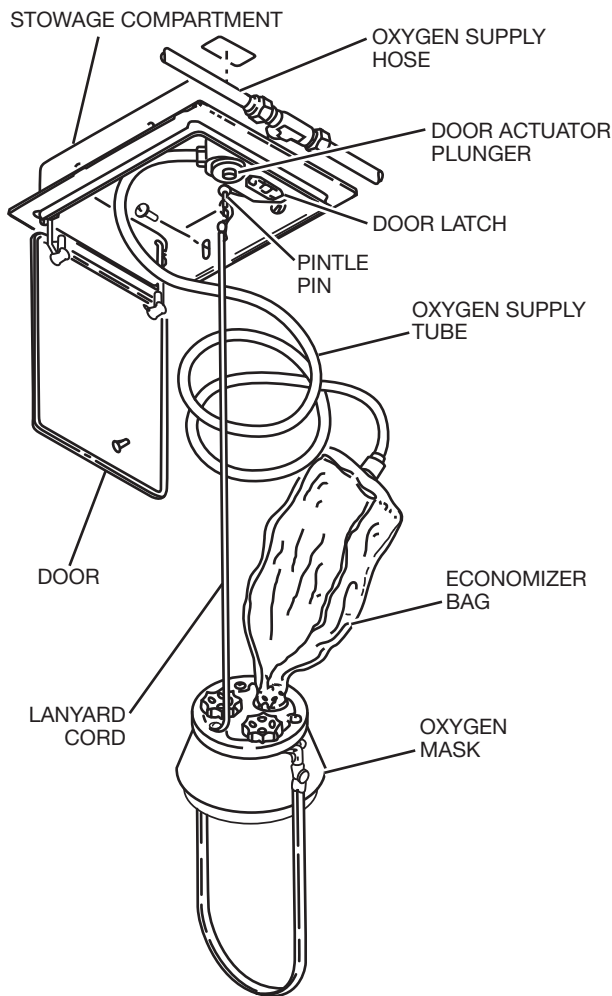


Figure 17-10. Passenger Oxygen Mask Installation

Walkaround Oxygen Bottle

A crew walkaround oxygen bottle with two oxygen masks and a shoulder strap (Figure 17-11) is standard equipment and is worn for extinguishing a cabin fire. One crew mask can be connected to the bottle. Passenger masks cannot be connected to the walkaround bottle. The bottle contains 11 cubic feet of oxygen and is charged to 1,800 psi.

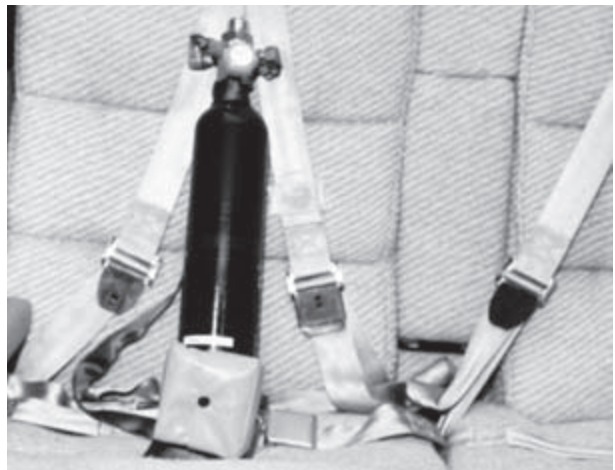


Figure 17-11. Walkaround Oxygen Bottle

If a cabin fire occurs, don the smoke goggles (Figure 17-12), and then disconnect the oxygen mask from the aircraft system. Connect the crew mask to the walkaround bottle to allow breathing if heavy smoke occurs.

Time of Useful Consciousness and Oxygen Duration

Table 17-1 provides the average period of useful consciousness (time from loss of pressure



Figure 17-12. Smoke Goggles

until loss of effective performance) at various altitudes. Tables 17-2 and 17-3 provide the expected oxygen duration using the standard mask and either the 49 cubic-foot bottle or the 76 cubic-foot bottle, respectively.

LIMITATIONS

Oxygen Mask

Prior to flight, check, adjust, and properly stow all crew masks.

Crew oxygen masks are not approved for cabin altitudes greater than 40,000 feet. Passenger oxygen masks are not approved for cabin altitudes greater than 25,000 feet.

Remove all headsets and/or hats prior to donning the oxygen masks.

NOTE

Headsets, eyeglasses, or hats worn by the crew may interfere with the quick-donning capabilities of the oxygen masks.

When using the crew oxygen mask, ensure that the audio amplifier AUTO SEL COMM switch(es) is positioned to SPKR in order to receive audio from the communication radios.

Oxygen System

Service the oxygen system with aviator's breathing oxygen per MIL-0-27210.

AIRCRAFT DOORS

The aircraft have up to eight doors used by flight and ground crew:

- Cabin door
- Emergency exit
- Two nose section doors
- Aft baggage compartment door
- Toilet service door
- Single-point pressure refueling (SPPR) door
- Tailcone compartment door

CABIN DOOR

The one-piece cabin door is hinged at the bottom and opens outward (Figure 17-13). An overcenter lock, which holds the door open, is released either by the handrail or by the actuating handle in the cabin. The door is counterbalanced by springs for easy opening or closing. The door is secured closed with 10 locking pins—four on each side, two at the top edge—and a hinge at the bottom.

Four microswitches and a handle proximity switch connect to the CAB DOOR UNLOCKED annunciator. Eleven sight windows are provided to verify the locking pin position and the over-center mechanism when the door is locked.

When the door is locked, the lower aft locking pin presses a plunger, which opens a valve and allows service air to inflate the pneumatic primary seal thereby preventing cabin pressure loss. A secondary compression seal holds cabin pressure up to an aircraft altitude of 51,000 feet.

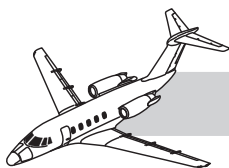


Table 17-1. AVERAGE TIME OF USEFUL CONSCIOUSNESS OR AVERAGE EFFECTIVE PERFORMANCE TIME

ALTITUDE	TIME
15,000 –18,000	30 minutes or more
22,000	5–10 minutes
25,000	3–5 minutes
28,000	2 1/2–3 minutes
30,000	1–2 minutes
35,000	30–60 seconds
40,000	15–20 seconds
45,000	9–15 seconds
51,000	Less than 9 seconds

NOTE:

EFFECTIVE PERFORMANCE TIME IS SIGNIFICANTLY REDUCED WITH EXPLOSIVE DECOMPRESSION

Table 17-2. OXYGEN DURATION TIME (49-CUBIC-FOOT CYLINDER)

AVAILABLE TIME IN MINUTES									
CABIN ALTITUDE	1 COCKPIT	2 COCKPIT	2 COCKPIT 2 CABIN	2 COCKPIT 4 CABIN	2 COCKPIT 6 CABIN	2 COCKPIT 8 CABIN	2 COCKPIT 10 CABIN	2 COCKPIT 12 CABIN	2 COCKPIT 13 CABIN
8,000	676	338	109	65	46	36	29	25	23
10,000	713	357	111	66	47	36	29	25	23
15,000	803	402	115	67	47	37	30	25	23
20,000	917	459	119	68	48	37	30	25	23
25,000	428	214	92	58	43	34	28	24	22
30,000	558	279	–	–	–	–	–	–	–
35,000	755	378	–	–	–	–	–	–	–
37,000	917	459	–	–	–	–	–	–	–
39,000	1,070	535	–	–	–	–	–	–	–

Table 17-3. OXYGEN DURATION TIME (76-CUBIC-FOOT CYLINDER)

AVAILABLE TIME IN MINUTES									
CABIN ALTITUDE	1 COCKPIT	2 COCKPIT	2 COCKPIT 2 CABIN	2 COCKPIT 4 CABIN	2 COCKPIT 6 CABIN	2 COCKPIT 8 CABIN	2 COCKPIT 10 CABIN	2 COCKPIT 12 CABIN	2 COCKPIT 13 CABIN
8,000	1,040	520	167	99	71	55	45	38	35
10,000	1,098	549	170	100	71	55	45	38	35
15,000	1,235	617	176	102	72	56	45	39	36
20,000	1,412	706	183	105	73	56	46	39	36
25,000	659	344	141	89	65	52	42	37	34
30,000	859	429	–	–	–	–	–	–	–
35,000	1,162	581	–	–	–	–	–	–	–
37,000	1,412	706	–	–	–	–	–	–	–
39,000	1,647	823	–	–	–	–	–	–	–

NOTES:

COCKPIT MASKS ARE ASSUMED TO BE AT THE NORMAL SETTING UP TO 20,000 FEET WITH A RESPIRATORY RATE OF 10 LITERS PER MINUTE—BODY TEMPERATURE PRESSURE SATURATED. ABOVE 20,000 FEET, THE MASKS ARE ASSUMED TO BE AT THE 100% SETTING.

PASSENGER OXYGEN MASKS ARE APPROVED FOR OPERATION AT CABIN ALTITUDE OF 25,000 FEET AND BELOW.



Figure 17-13. Cabin Door

A DOOR TEST switch, if installed, is on the interior door frame and is used for testing four perimeter microswitches (Figure 17-14).



Figure 17-14. DOOR TEST Switch

Opening the Door from Outside

If the cabin door is locked, insert the key into the lock below the outside cabin door handle, and then rotate the key 90° clockwise. Pull the flush-mounted handle out by the finger hole in the small end, and then rotate the handle 90° clockwise to unlatch the door. Pull the door outward and down. When the door reaches its lowest point, step on the lower step, and

then press the handrail down until the linkage goes overcenter.

If the door is held closed only by the precatch, or if the precatch engages when the door unlatches, release the precatch by pressing the precatch release button just forward of the door (Figure 17-15).

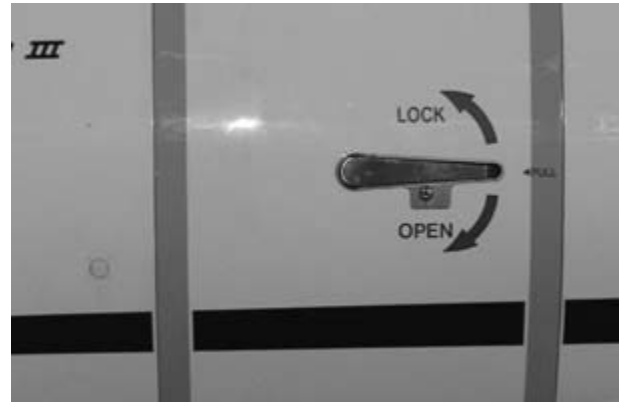


Figure 17-15. Outside Door Latch

Opening the Door from Inside

The flush-mounted, inside cabin door handle is in the lower door step. (The lower step is at the top of the door when the door is closed). To open the door from the inside, push the button, pull out on the small end of the handle, and then rotate the handle 90° counterclockwise to unlatch the door (Figure 17-16). The locking mechanism retracts the locking pins from the door-frame sockets. If the door does not open after being unlatched, then the precatch in the forward upper socket may be extended. Pull the red knob forward of the door to release the precatch.

If electrical power is on the aircraft when the locking pins retract, the warning switches activate, completing an electrical ground and illuminating the CAB DOOR UNLOCKED annunciator.

When the lower aft locking pin retracts, the valve allowing service bleed air to inflate the door seal closes and the door seal will deflate (allow 3–5 seconds for seal deflation). Pushing on the upper portion of the door starts the



Figure 17-16. Inside Door Latch

door outward and downward and moves the door actuating lever (handle) downward. The counterbalance mechanism prevents the door from free-falling. An overcenter locking linkage and two telescoping support struts provide solid footing for cabin entry and exit.

Closing the Door from Inside

To close the entrance door from inside the cabin, lift the handle, which raises the door to the closed position (Figure 17-17). As the door raises, the counterbalance mechanism assists the closing effort and the handrail folds toward the door.

To close and lock the door, pull the PULL handle on the left side of the lower step with sufficient force to engage the precatch. Push the button to displace the inside cabin door handle, and then rotate the handle 90° clockwise to latch the door.

The locking mechanism extends the locking pins into the door frame sockets. If main DC power is on the aircraft when the locking pins extend and the proximity switch opens, then the warning switch opens the electrical circuit and extinguishes the CAB DOOR UNLOCKED annunciator.



Figure 17-17. Inside Door Handle

NOTE

If any warning switches do not agree that the door is locked when the aircraft is on the ground, then the CAB DOOR UNLOCKED annunciator flashes.

At the same time, the lower aft locking pin opens the door seal valve to permit inflation of the primary door seal when service bleed air is available. Check the locking handle linkage for proper position by the alignment of the marks on the linkage and the sight window near the locking handle. When the marks align, the linkage is overcenter and locked.

NOTE

If the cabin door is not closed and locked when the throttles are advanced, the NO TAKEOFF annunciator illuminates, and an aural warning sounds.

Closing Door from the Outside

To close the entrance door from the outside, stand on the forward side of the door, and then lift the handrail to release the actuating lever linkage from overcenter. With the aid of the counterbalance mechanism, lift the door to



the closed position with enough force to engage the precatch. Pull the outside cabin door handle outward by the finger hole, and then rotate the handle 90° counterclockwise to latch the door. If desired, lock the door handle by rotating the key 90° counterclockwise with the handle in the stowed position.

WARNING

Do not stand directly in front of the lower step when releasing the door from the locked down position. The door may spring upward, striking the legs.

AIRCRAFT SERVICING

FUEL

The aircraft can use a variety of fuels. Approved fuels are:

- Jet A
- Jet A-1
- Jet B
- JP-4
- JP-5
- JP-8

Different fuels have different specific gravity that effect operation of the engine electronic fuel control system. Refer to the FAA-approved *AFM* and the *Garrett Engine Maintenance Manual* for computer settings for optimum engine acceleration.

Use of anti-ice additive in the aircraft is not necessary to prevent fuel system icing since the engines have fuel heaters. However, the approved anti-ice additive is biocidal, which controls bacteria and fungi. Aircraft being serviced without anti-ice additive on a continual basis should have a program established for controlling microorganisms of bacteria

and fungi. Refer to the *AFM* for servicing information.

The aircraft are fueled through a SPPR receptacle aft of the right wing. Wing tanks and the aft fuselage tank are serviced simultaneously.

An overwing port is in each wing for refueling if pressure refuel facilities are unavailable. When refueling through the overwing ports, refuel the fuselage tank from the left wing tank. At a predetermined wing tank level (approximately 175 gallons), the left boost pump can be used to transfer fuel to the fuselage tank. A switch under a left wingtip access door controls the boost pump. The wing tank is topped off after the fuselage tank fills. Procedures for transferring fuel to the fuselage tank are displayed on a placard under the left wingtip.

To control microorganisms in the fuel tanks if not using fuel with anti-ice additive start refueling while simultaneously applying the fuel additive. Direct the additive into the flowing fuel stream for proper mixing. Start the additive flow after fuel flow begins, and stop before fuel flow ends.

- Do not allow concentrated additive to contact the coated interior of the fuel tank or painted surfaces of the aircraft.
- Do not use less than 20 fluid ounces of additive per 260 gallons of fuel or more than 20 fluid ounces per 104 gallons of fuel.
- Insufficient additive concentrations may result in growth of damaging microorganisms in the fuel system. Excessive additive may cause fuel tank damage or erroneous fuel quantity indications.
- When refueling, do not operate radios, radar, or other electronic equipment.
- Ensure that the fuel truck and aircraft are grounded and that a bond wire is connected from the truck to the aircraft. A fuel ground plug attachment is under each wingtip.



It is not necessary to maintain fuel balance during refueling. Maximum asymmetric fuel differential for flight is 200 pounds; however, the aircraft can accommodate as much as 800 pounds of differential in an emergency.

OIL

Each engine has an oil filler under an access door on the upper outboard side of the engine. The right engine is filled directly into the oil tank and the left engine through a fill tube connected to the tank on the opposite side of the engine. The oil quantity sight gauge is inside a small door on the right side of the engine. To obtain an accurate check of the oil quantity, check the oil when the engine is hot, within one hour of engine shut down.

WARNING

Persons who handle engine oil are advised to minimize skin contact with used oil, and promptly remove any used oil from their skin. A laboratory study, while not conclusive, found substances which may cause cancer in humans. Thoroughly wash used oil off skin as soon as possible with soap and water. Do not use kerosene, thinners or solvents to remove used engine oil. If waterless hand cleaner is used, always apply skin cream after using.

The engines must be serviced only with the approved oils listed in the *AFM*. The type of oil used in each aircraft is indicated in the engine logbook as well as displayed on a placard inside the filler access door.

The onboard APU (if installed) has an oil filler cap and a sight glass on the reduction drive assembly. Either the dipstick or the sight glass is used for checking the oil level and is readily accessible for easy serviceability. Service the APU only with the approved oils listed in the FAA-approved *AFM*.

Each pneumatic air conditioner assembly is equipped with a sight glass and a filler plug. The sight glasses and filler plugs are on the left side of the PACs and are readily accessible for easy serviceability. The pneumatic air conditioner assembly must be serviced with only the approved oils listed in the *Operating Manual*.

HYDRAULIC

Servicing the hydraulic reservoir requires equipment capable of delivering hydraulic fluid under pressure and is performed by maintenance personnel. Service the reservoir with only the fluid listed in the *Operating Manual*.

OXYGEN

The oxygen filler valve is inside a small access door in the nose compartment. Oxygen servicing must be done by maintenance personnel using only the approved breathing oxygen listed in the *AFM*. Reference the cockpit gauge while servicing to prevent overfill.

ALCOHOL

An alcohol reservoir (Citation III/VI only) is located on the forward pressure bulkhead. The filler plug on the reservoir should be removed and alcohol added to bring the fluid level up to the neck of the filler plug. Filling to above the sight gauge provides a reserve supply of alcohol to perform preflight or operational checks without replenishing the reservoir.

FIRE BOTTLES

Underserviced fire bottles must be exchanged by authorized maintenance facilities. Low bottle service is indicated by illumination of the FIRE EXT BOTTLE LOW annunciator on the annunciator panel.

GEAR AND BRAKE PNEUMATIC SYSTEM

Service the emergency gear and brake bottles when the pressure gauge reads below 1,800



psi. Maintenance personnel must perform the servicing with high pressure nitrogen and refill the bottles to 2,050 psi. Servicing is accomplished through a charging valve on the bottles which are in the nose compartment on the left side of the forward pressure bulkhead.

TIRES

The main gear tire pressure must be maintained at 168 psi, when the aircraft is on the ground (165 psi when the aircraft is on jacks). The nose tire pressure must be maintained at 140 psi when the aircraft is on the ground (138 psi when the aircraft is on jacks).

Since tire pressure decreases as the temperature drops, a slight overinflation can be used to compensate for cold weather. Main tires inflated at 21°C should be inflated 1.5 psi for each 6°C drop in temperature anticipated at the coldest airport of operation. Nose tires at 21°C should be overinflated only 0.5 psi for each 6°C anticipated drop in temperature.

TOILET

Citation III/VI

The flush toilet reservoir requires servicing when the liquid level becomes too low or when liquid appears to have incorrect chemical balance. To properly service the reservoir it must be removed from the toilet by disconnecting it and pulling it through the door in the front of the cabinet. Instructions for removing and servicing the reservoir are found in Chapter 2 and 38 of the *AMM*. Servicing the reservoir requires the addition of the proper mixture of water and chemical (three ounces of chemical per quart of water) to the reservoir. It takes more than two quarts of liquid if the reservoir is empty.

If outside temperatures are below freezing and the aircraft is kept in an unheated hanger, add antifreeze to both the reservoir and the waste container.

Citation VII

An optional, externally serviceable toilet is available. The service port is on the lower aft side of the fuselage, directly behind the left inboard flap panel (Figure 17-18).



Figure 17-18. Toilet Service Port

Service the toilet during routine ground maintenance after any usage. A toilet ground service cart is required for servicing.

Maximum capacity of the toilet tank is 4.0 gallons and an overfill light is provided.

To ensure operation of the toilet recirculation system during freezing weather, add an ethylene glycol base antifreeze with an antifoam agent to the flush toilet.

AIRCRAFT CLEANING AND CARE

OXYGEN MASKS

The crew masks are permanent masks which contain a microphone for radio transmissions. The passenger masks are oro-nasal type, which conform around the mouth and nose area. All masks can be cleaned with alcohol. Do not



allow the solution to enter the microphone or electrical connections. Apply talcum powder to external surfaces of the passenger mask rubber face piece.

PAINTED SURFACES

The painted exterior surfaces of a newly painted aircraft require an initial curing period which may be as long as 90 days after the finish is applied. During this curing period precautions must be taken to avoid damaging the finish or interfering with the curing process. The finish must be cleaned only by washing with clean water and mild soap, followed by a rinse water and drying with cloths or cham- ois. Do not use polish or wax, which would exclude air from the surface, during this 90-day curing period. Do not rub or buff the finish and avoid flying through rain, hail, or sleet.

To help prevent development of corrosion, particularly filiform corrosion, the aircraft must be spray washed at least every two to three weeks and waxed with a good automotive grade of water repellent wax. This is especially true in warm, damp, and salty environments. A heavier coating of wax on the leading edge of the wing and tail and on the engine nose cone helps reduce abrasions encountered in these areas.

ENGINES

Clean the engine compartments using a suitable solvent. Most efficient cleaning is done using a spray-type cleaner. Before spray cleaning, ensure protection is afforded for components which might be adversely affected by the solvent. Refer to the *Airplane Maintenance Manual* for proper lubrication of components after engine cleaning.

INTERIOR CARE

To remove dust and loose dirt from the upholstery, headliner, and carpet, clean the interior regularly with a vacuum cleaner.

Blot up any spilled liquid promptly with cleansing tissue or rags. Do not pat the spot;

press the blotting material firmly and hold it for several seconds. Continue blotting until no more liquid is taken up. Scrape off sticky materials with a dull knife, then spot-clean the area.

Oily spots can be cleaned with household spot removers, used sparingly. Before using any solvent, read the instructions on the container and test it on an obscure place on the fabric to be cleaned. Never saturate the fabric with a volatile solvent; it may damage the padding and backing materials.

WARNING

Use all cleaning agents in accordance with the manufacturer's recommendations. The use of toxic or flammable cleaning agents is discouraged. If these cleaning agents are used, ensure adequate ventilation is provided to prevent harm to the user, and/or damage to the aircraft.

Clean soiled upholstery and carpet with normal cleaning materials used according to the manufacturer's instructions. To ensure that the cleaning materials or chemicals will not harm the fabric or leather to be cleaned, test it on a hidden part of the material before proceeding.

The plastic trim, instrument panel, and control knobs need only be wiped with a damp cloth. Oil and grease on the control wheel and control knobs can be removed with a cloth moistened with kerosene. Volatile solvents, such as those mentioned in the care of the windshield, must never be used since they soften and craze the acrylic.

WINDOWS AND WINDSHIELDS

The Citation 650 series aircraft are equipped with either clear, stretched acrylic or glass windshields. On aircraft equipped with acrylic windshields, all other cockpit and cabin windows are acrylic, as well. On aircraft equipped with glass windshields, both forward cockpit side windows are also glass, while all other cockpit and cabin windows are acrylic. All



windshields and windows should be kept clean at all times. Acrylic windshields and windows should be waxed with a coat of good quality commercial wax. Do not wax glass windshields and windows.

To prevent scratches wash the windshields and windows carefully with plenty of soap and water, using the palm of the hand to feel and dislodge dirt and mud. A soft cloth, chamois or sponge may be used, but only to carry water to the surface. Rinse thoroughly, and then dry with a clean, moist chamois. Rubbing the surface of acrylic windows and windshields with a dry cloth builds up an electrostatic charge which attracts dust particles in the air. Wiping with a moist chamois removes both the dust and the charge.

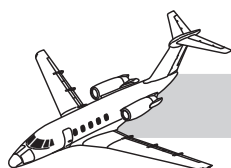
Remove oil and grease with a cloth lightly moistened with kerosene. Never use gasoline, benzine, acetone, carbon tetrachloride, fire-extinguisher fluid, lacquer thinner, or glass cleaner. These materials soften the plastic on the windows and can cause it to craze. Do not use strong chemicals and abrasive materials on the windshields and cockpit side windows. They can scratch and otherwise damage the repellent surface.

After removing dirt and grease, if the plastic surfaces are not badly scratched, they should be waxed with a good grade of commercial wax. The wax fills in minor scratches and helps prevent further scratching. Apply a thin, even coat of wax and bring it to a high polish by rubbing lightly with a clean, dry soft flannel cloth. Do not use a power buffer; the heat generated by the buffing pad may soften the acrylic. Do not use a canvas cover on the windshield unless freezing rain or sleet is anticipated. Canvas covers can scratch the repellent coating on the surface.



QUESTIONS

1. The pilot must ensure oxygen is available prior to flight by:
 - A. Reading the oxygen quantity gauge in the cockpit
 - B. Reading the oxygen pressure gauge in the nose compartment during the preflight inspection
 - C. Placing the oxygen mask regulator to EMER and hearing the rush of air from the mask
 - D. Observing the green band in the crew mask supply line
2. With the standard mask regulator positioned at 100%:
 - A. The pilot receives 100% oxygen
 - B. The pilot receives an increasing mixture of oxygen and cabin air to a maximum of 100% at 30,000 feet
 - C. An automatic, positive pressure schedule above 37,000 feet is provided
 - D. Both A and C are correct
3. In order to receive oxygen through the passenger mask:
 - A. The PASS OXY mask regulator must be positioned to ON
 - B. The lanyard must be pulled to remove the pintle pin from the valve
 - C. The economizer bag must be fully inflated
 - D. The lanyard and pintle pin must be inserted into the valve
4. With the use of Table 17-1, the time of useful consciousness in the event of a rapid decompression at 45,000 feet is:
 - A. 9–15 seconds
 - B. 30–60 seconds
 - C. 3–5 minutes
 - D. 30 minutes or more
5. When using the oxygen mask, in order to receive communications, the audio amplifier AUTO SEL switch must be positioned to:
 - A. EMER COMM 1
 - B. HDPH (headphone)
 - C. SPKR
 - D. COMM
6. If normal DC power is lost:
 - A. The passenger masks deploy immediately
 - B. The passenger masks cannot be deployed
 - C. Automatic deployment of the passenger masks does not occur
 - D. The oxygen gauge on the pilot panel is inoperative
7. If the PASS OXY control is placed in OFF:
 - A. Oxygen is not available to the crew or passenger masks
 - B. The passenger masks deploy automatically, but oxygen is not available
 - C. The passengers receive oxygen above a cabin altitude of 8,000 feet anyway
 - D. Oxygen is available to the crew masks
8. If the PASS OXY control is ON:
 - A. Oxygen is available to the crew masks only
 - B. The passenger masks do not deploy until a cabin pressure altitude of 13,500 feet is reached
 - C. The passenger masks deploy
 - D. Oxygen is available to the passengers only while above 8,000 feet cabin altitude

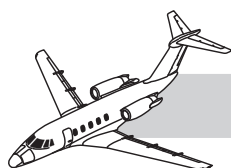


CHAPTER 18

MANEUVERS AND PROCEDURES

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CHAPTER 18

MANEUVERS AND PROCEDURES



OPERATING TECHNIQUES

PREFLIGHT AND TAXI PROCEDURES

First Flight of the Day

A thorough preflight inspection shall be conducted in accordance with the FAA-approved *Airplane Flight Manual (AFM)*. The format shall emphasize the safety of flight items.

Subsequent Flights

A walkaround shall be performed in order to conduct a visual preflight inspection to detect any significant changes since the aircraft was parked. The inspection shall include checks

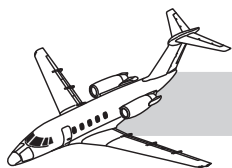
for removal of chocks, engine plugs, and pitot covers, condition of tires, nose steering mechanism, fluid leaks, and security of fuel caps. A check should be made to ensure that there is adequate fuel on board for the next flight.

Before Engine Start/Before Taxi

System checks shall be conducted in accordance with the FAA-approved *AFM* for all aircraft systems required for the flight. Flight control, communications, navigation, and electronic checks should be conducted prior to taxiing, if feasible.

Taxi

Systems checks carried out during taxiing should be kept to a minimum. The primary concern of the crew must be safe ground movement.



TAKEOFF PROCEDURES

General

The pilot flying the aircraft advances the throttles, slowly at first, to allow the engines to accelerate, then more rapidly to the computed takeoff power setting. The pilot monitoring guards the throttles and makes the final setting at approximately 60 KIAS. In addition, the pilot monitoring makes the following airspeed calls:

1. Airspeed alive (Initial airspeed indications on both instruments)
2. 80 knots (cross-checked)
3. V_1
4. V_R (call "rotate")
5. V_2

Takeoff Aborted

If an abnormal situation, annunciator, system failure, etc., should occur prior to V_1 , whichever pilot first notices the malfunction should immediately call "Abort."

If the decision is made to abort the takeoff prior to reaching V_1 , the following procedures from the checklist should be used:

1. Directional Control..... MAINTAIN
2. Brakes APPLY MAXIMUM EFFORT
3. Throttles IDLE
4. Spoilers EXTEND
5. Thrust Reverser(s) DEPLOY
 - a. Verify illumination of thrust reversers ARM, UNLOCK, and DEPLOY lights.
 - b. Verify illumination of RUDDER BIAS light.
 - c. Reverse Power (use as required if no fire)

Normal Takeoff

At V_R , the pilot rotates the aircraft to a 10° nose-up attitude on the attitude director indi-

cator and, when a positive rate of climb is indicated, retracts the gear. As the airspeed increases through a minimum of $V_2 + 25$ knots (V_{FR}), retract the flaps. Continue to accelerate to normal climb speed, and complete the after takeoff-climb checklist.

Engine Failure at or after V_1

If an engine fails at or after V_1 , the takeoff is normally continued. At V_R , rotate the nose of the aircraft to 10° , and raise the landing gear when a positive rate of climb has been established. Maintain V_2 until 1,500 feet AGL or obstacle clearance has been achieved, whichever is higher. Level off and accelerate to V_{ENR} . As the airspeed reaches $V_2 + 20$ or 25 (depending on model), retract the flaps. When V_{ENR} (165 KIAS) is obtained, reduce power to maximum continuous and climb at V_{ENR} to the minimum safe altitude (MSA) or assigned altitude. When time and cockpit duties permit, complete the appropriate emergency procedures checklist and the after takeoff-climb checklist.

NOTE

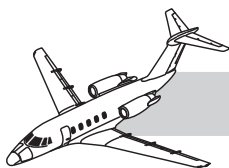
Do not let an emergency distract you from flying the aircraft. Wait until you are safely airborne and clear of obstacles before taking care of the emergency and the after-takeoff-climb checklist.

NOTE

During third segment acceleration, if after five minutes, enroute climb speed (V_{ENR}) has not been attained, hold altitude, reduce power to maximum continuous thrust and continue accelerating to the enroute climb speed (V_{ENR}).

PERFORMANCE

The Citation 650 series aircraft is certified under FAR Part 25. The following discussion is of use in understanding the capabilities of the aircraft.



Holding Speed

The manufacturer's published holding speeds are between 170 and 210 KIAS, depending upon weight and conditions. If fuel is critical, the speed may be reduced to 0.6 on the angle-of-attack indicator. This provides the maximum endurance.

Hydroplaning Speed

The formula used to determine the speed at which a tire is likely to hydroplane after touchdown on a wet runway is stated as:

$$\text{Hydroplane speed} = 7.7 \text{ to } 9 \times (\text{square root of the tire pressure})$$

Takeoff Limitations (Flaps 7° and 20°)

Figure 18-1 shows a typical takeoff data card.

The takeoff weight is limited by the most restrictive of the following requirements:

- Maximum certified takeoff weight (structural)
- Maximum takeoff weight permitted by climb requirements
- Maximum takeoff weight permitted by takeoff field length, which meets two requirements in the event of an engine failure at V_1 . It ensures that the rejected takeoff can be completed within the existing runway. Also, it allows for the takeoff to be continued, ensuring that the aircraft reaches a height of 35 feet (Reference Zero) by the time it reaches the end of the takeoff distance.

These requirements are operating limitations and must be complied with. Additionally, obstacle clearance capability may be an actual physical necessity as well as a legal requirement and may further limit the takeoff weight.

The pilot should also consider the landing weight restrictions at the destination airport. The limited landing weight plus the expected fuel to be burned enroute may be more limiting than any restrictions at the departure airport, especially if the trip is of short duration.

FlightSafety international		CITATION	
TAKEOFF DATA			
T/O N_1		CLB N_1	
V_1	V_R	V_2	
V_{FR}	V_{ENR}	FLAPS	
CLEARANCE			
ARPT _____ ELEV _____ RWY _____			
ATIS _____ WIND _____ VIS _____			
CIG _____ TEMP/DP _____ / _____			
ALT _____ RMKS _____			
RWY LENGTH _____ RWY REQ'D _____			
ZFW _____ T.O. WT. _____			
EMERGENCY RETURN			
V_{REF} _____ V_{APP} _____ MSA _____			

Figure 18-1. Takeoff Data Card



T.O. N_1 —Fan setting for existing temperature and pressure altitude taken from the *AFM* or checklist; must be adjusted for anti-ice, if used.

CLB N_1 —Maximum fan setting for existing temperature and pressure altitude taken from the *AFM* or checklist; must be adjusted for anti-ice, if used.

V_1 —Takeoff decision speed computed from the *AFM* or checklist (may be adjusted for runway slope and use of anti-ice, but can never be greater than V_R)

V_R —Rotation Speed computed from the *AFM* or checklist

V_2 —Takeoff safety speed; this is the actual speed at 35 feet above the runway surface as demonstrated in flight with one engine inoperative. Essentially best angle of climb speed.

V_{FR} —Flap retracting speed ($V_2 + 25$ knots); also used as minimum final segment climb speed ($V_2 + 20$ for some models)

V_{ENR} —Single-engine enroute climb speed (always 165 knots). Essentially best rate of climb speed.

FLAPS—Flap setting for takeoff

CLEARANCE—Space used for copying clearance or any additional information required

ARPT—Name of airport or ICAO identifier

ELEV—Airport elevation or runway elevation if significantly different from airport elevation

RWY—Runway in use as reported by ATIS

ATIS—Identifying letter of Airport Terminal Information Service (current pertinent airport information)

WIND—Wind direction and speed as reported by ATIS

VIS—Visibility as reported by ATIS

CIG—Ceiling as reported by ATIS

TEMP/DP—Temperature/dew point of airport as reported by ATIS

ALT—Altimeter setting as reported by ATIS

RMKS—Any pertinent remarks as reported by ATIS

RWY LENGTH—Actual length of runway to be used for takeoff

RWY REQ'D—Actual takeoff field length obtained from *AFM* or checklist adjusted for current conditions

ZFW—Zero Fuel Weight (empty weight plus payload)

T.O. WT.—Takeoff Weight (ZFW plus takeoff fuel)

V_{REF} —Approach speed for landing flap schedule at the emergency return landing weight.

V_{APP} —Go-around speed adjusted for flap schedule at the emergency return landing weight. Obtained from the *AFM* or checklist

MSA—Minimum Safe (Sector) Altitude within 25 nautical mile radius of NAV facility upon which the procedure is predicated

Obstacle Clearance (Loss of Engine at V_1)

FAR Part 25 requires that the aircraft manufacturer display a takeoff flight path profile (Figure 18-2) beginning at reference zero and ending at 1,500 feet AGL. For obstacle clearance purposes, second segment climb charts show net takeoff climb gradients (actual or gross climb gradient reduced by 0.8%).

Enroute

The single-engine enroute climb gradient table allows the calculation of the maximum enroute altitude that the aircraft will maintain on one engine. The table displays the enroute net climb gradient (actual or gross climb gradient reduced by 1.1%).



CITATION 650 SERIES PILOT TRAINING MANUAL

	FIRST-SEGMENT CONFIGURATION	SECOND-SEGMENT CONFIGURATION	THIRD-SEGMENT CONFIGURATION
SPEED	V_2	V_2	V_2 ACCELERATING TO V_{ENR}
THRUST SETTING	TAKEOFF/GO AROUND	TAKEOFF/GO AROUND	TAKEOFF/GO AROUND (FOR 5 MINUTES, THEN MAXIMUM CONTINUOUS SINGLE ENGINE)
FLAP POSITION	7° OR 20°	SAME AS FIRST SEGMENT	7° OR 20° TRANSITING TO UP
GEAR POSITION	DOWN TRANSITING TO UP	UP	UP
REQUIRED GRADIENT	POSITIVE	2.4% GROSS/1.6% NET	NONE

NOTE:
THE SECOND SEGMENT IS THE LIMITING SEGMENT AS FAR AS MAXIMUM TAKEOFF WEIGHT PERMITTED BY CLIMB IS CONCERNED.

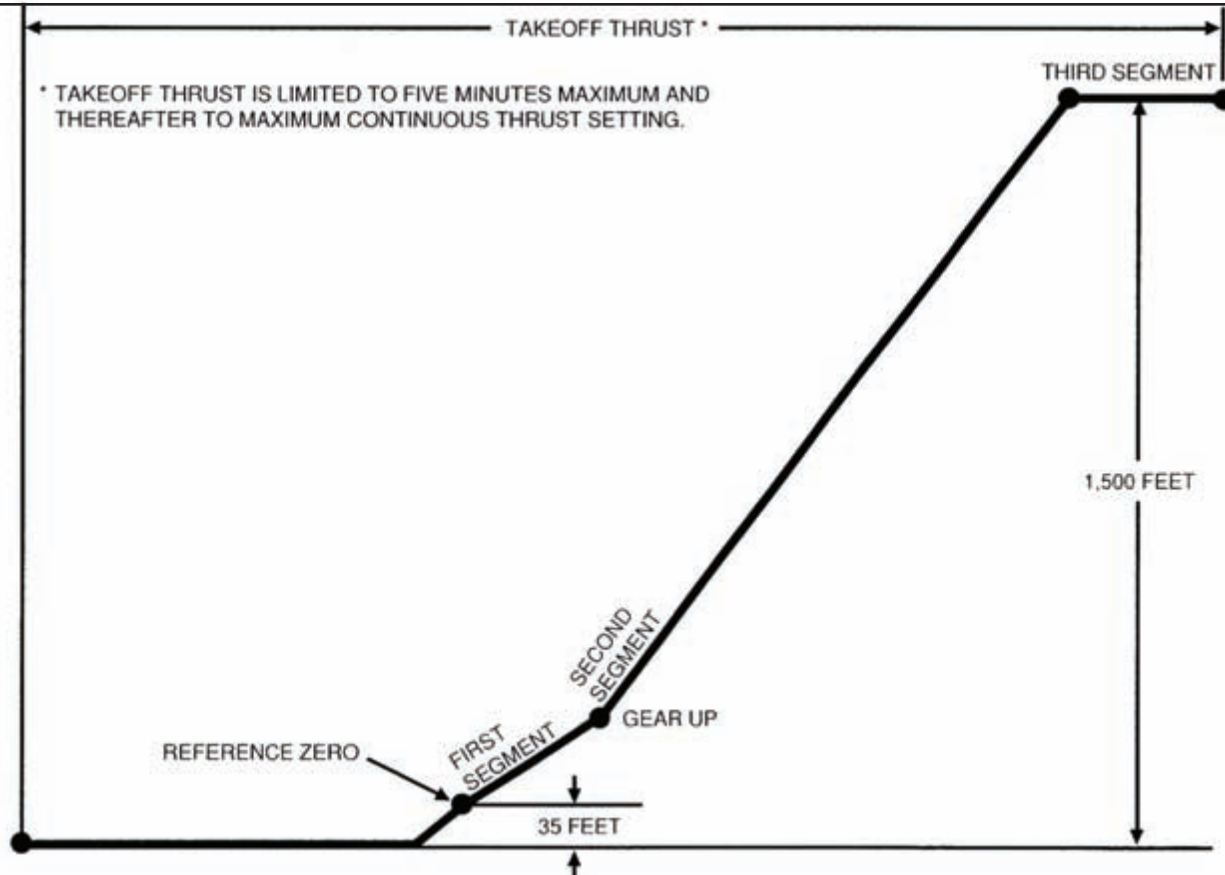


Figure 18-2. Takeoff Flight Path Profile



Landing Limitations

Figure 18-3 shows a typical landing data card.

The landing weight is limited by the most restrictive of the following requirements:

- Maximum certified landing weight (structural)
- Maximum landing weight permitted by climb requirements
- Maximum landing weight permitted by brake energy limits
- Maximum landing weight permitted by field length (both landing and subsequent takeoff)

The landing weight permitted by climb requirements is based upon the most limiting of the approach or landing climb configurations (Table 18-1). Since the approach climb is based upon one engine operating, it is the more limiting of the two. Therefore, the maximum landing weight permitted by the climb table lists the landing weight as limited by the approach climb.

The landing distance table provides the horizontal distance necessary to land and come to a complete stop from a point over the runway threshold at V_{REF} ($1.3 V_{SO}$). The landing is preceded by a 3° glidepath down to the 50-foot point, where the engine thrust is reduced to idle.

FlightSafety [®] international		CITATION
LANDING DATA		
V_{REF}	V_{APP}	
GA N_1	RWY REQ'D	
CLEARANCE		
ARPT _____ ELEV _____ RWY _____		
ATIS _____ WIND _____ VIS _____		
CIG _____ TEMP/DP _____ / _____		
ALT _____ RMKS _____		
ZFW _____ LDG WT. _____		

Figure 18-3. Landing Data Card

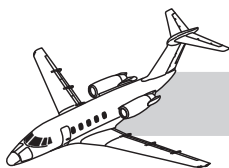


Table 18-1. APPROACH AND LANDING CLIMB CONFIGURATIONS

	APPROACH CLIMB CONFIGURATION	LANDING CLIMB CONFIGURATION
SPEED	V_{APP} ($1.3 V_{S1}$) (APPROACH CLIMB SPEED)	V_{REF} ($1.3 V_{S0}$) (LANDING CLIMB SPEED)
ENGINES	ONE	TWO
THRUST SETTING	TAKEOFF/GO AROUND	TAKEOFF/GO AROUND
FLAP POSITION	APPROACH (7° OR 20°)	LAND (20° OR FULL)
GEAR POSITION	UP	DOWN
REQUIRED CLIMB GRADIENT	2.1% GROSS	3.2% GROSS

V_{REF} —Approach speed for landing flap schedule (adjusted for partial or no flaps) obtained from the *AFM* or checklist. Based upon weight.

V_{APP} —Go-around speed adjusted for flap schedule at the landing weight. Obtained from the *AFM* or checklist

$GA N_1$ —Go-around fan setting obtained from the *AFM* or checklist based upon OAT

RWY REQ'D—Landing field length adjusted for aircraft configuration, landing weight and runway conditions.

CLEARANCE—Space for routes, arrival, and approach or other pertinent information obtained from ATIS or ATC.

ARPT—Name of destination airport or ICAO identifier

ELEV—Destination airport elevation or runway elevation if significantly different

RWY—Runway in use at destination airport

ATIS—Identifying letter of Airport Terminal Information Service (current pertinent airport information)

WIND—Wind reported at destination airport

VIS—Visibility as reported at destination airport

CIG—Ceiling as reported at destination airport

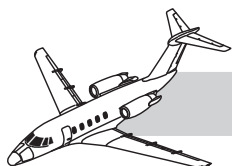
TEMP/DP—Temperature/dew point at destination airport

ALT—Current altimeter setting at destination airport

RMKS—Pertinent remarks as stated in ATIS at destination airport or by ATC

ZFW—Zero Fuel Weight (empty weight plus payload)

LDG WT—Landing weight (ZFW plus remaining fuel)

**Table 18-2. MINIMUM MANEUVERING SPEEDS**

FLAP CONFIGURATION	MINIMUM SPEEDS
CLEAN	$V_{REF} + 30$ KNOTS
FLAPS 20°	$V_{REF} + 20$ KNOTS
FLAPS FULL	$V_{REF} + 10$ KNOTS

Minimum Maneuvering Speed

This speed is the minimum speed that provides an adequate margin above stall while maneuvering the aircraft. Table 18-2 lists the Citation 650 series minimum maneuvering speeds in relationship to a full-flap V_{REF} .

FINAL APPROACH PROCEDURES

NOTE

The following are suggested procedures.

Flight Deck Discipline

Good operating practices are essential for precise execution of approach procedures, whether on instruments or visual. By constantly maintaining an awareness of the progress along the approach profile, the crew provides for an orderly transition to the landing runway; cross-checking must be thorough and continuous.

Approach planning begins sufficiently in advance of the approach, with a review of the approach charts and attention given to alternative courses of action.

Flight information redundancy improves the ability to cross-check, which in turn provides for a continuous validation of one information source against another. It also decreases the affect of over-concentration on a single instrument display.

The cross-check on final approach is, therefore, enhanced by tuning both pilot navigation aids to the same frequencies.

Scan Transfer

The transfer from instruments to visual flight differs with the approach being flown.

Noncoupled approaches:

- The pilot flying remains on instruments. When reaching DA or MDA and being advised of continuous visual reference, he progressively adjusts his scan to visual flight, announces “I am visual” and lands.
- The pilot monitoring, when approaching DA or MDA, adjusts his scan pattern to include outside visual cues. When the pilot flying announces that he is “visual” the pilot monitoring assumes the responsibility for monitoring the instruments and provides continuous advice of warning flags and deviations from approach tolerances (sink rate, airspeed, glide-slope, and localizer) to touchdown.

Coupled approaches:

- The pilot flying adjusts his scan pattern to include outside visual cues. When reaching DA or MDA and having assured himself of continuous visual reference, he announces, “I am visual” and lands.
- The pilot monitoring concentrates on instruments to touchdown, advising of warning flags and deviations from approach tolerances.

STANDARD CALLOUTS (IFR AND VFR)

Refer to Table 18-3 for a list of the standard callouts.

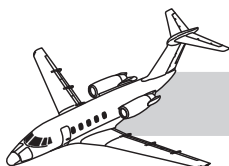


Table 18-3. STANDARD CALLOUTS

LOCATION	CONDITION	CALLOUT
TAKEOFF	• COMPUTED N_1 SET • ENGINE INSTRUMENTS NORMAL • ANNUNCIATOR PANEL LIGHTS NORMAL	
	BOTH AIRSPEED INDICATORS INDICATING 60 KIAS	"60 KNOTS"
	BOTH AIRSPEED INDICATORS INDICATING 80 KIAS	"80 KNOTS"
	AIRSPEED INDICATORS AT COMPUTED V_1	" V_1 "
	AIRSPEED INDICATORS AT COMPUTED V_R	"ROTATE"
	AIRSPEED INDICATORS AT COMPUTER V_2	" V_2 "
DEPARTURE/ ENROUTE/ APPROACH	PRIOR TO INTERCEPTING AN ASSIGNED COURSE	"COURSE ALIVE"
	1,000 FEET PRIOR TO LEVEL OFF	STATE ALTITUDE LEAVING AND ASSIGNED LEVEL-OFF ALTITUDE
CLIMB AND DESCENT	APPROACHING TRANSITION ALTITUDE/LEVEL (IFR AND VFR)	"TRANSITION ALTITUDE/LEVEL ALTIMETERS SET"
	1,000 FEET ABOVE/BELOW ASSIGNED ALTITUDE (IFR)	STATE ALTITUDE LEAVING AND ASSIGNED LEVEL-OFF ALTITUDE
FINAL (IFR)	AT FINAL APPROACH FIX	"(FIX) ALTIMETERS AND INSTRUMENTS CHECK" ¹
	500 FEET ABOVE MINIMUMS	"500 ABOVE MINIMUMS"
	100 FEET ABOVE MINIMUMS	"100 ABOVE MINIMUMS"
	VISUAL REFERENCE REQUIRED BY FAR 91.175(C) IS CONTINUOUSLY ESTABLISHED ²	"RUNWAY AT (CLOCK POSITION)," OR "APPROACH LIGHTS AT (CLOCK POSITION)"
	AFTER PILOT FLYING REPORTS "VISUAL," PILOT MONITORING REVERTS TO INSTRUMENTS AND CALLOUTS	" V_{REF} + AIRSPEEDS"
	"	"SINK (RATE OF DESCENT)"
	"	"ON," "ABOVE," OR "BELOW GLIDE SLOPE," IF AVAILABLE
	AT DA (DECISION ALTITUDE)	"MINIMUMS, RUNWAY NOT IN SIGHT," OR "MINIMUMS, RUNWAY AT (CLOCK POSITION)" OR "MINIMUMS, APPROACH LIGHTS AT (CLOCK POSITION)"
	AT MDA (MINIMUM DESCENT ALTITUDE)	"MINIMUMS"
	AT MAP (MISSED-APPROACH POINT)	"MISSED-APPROACH POINT, RUNWAY NOT IN SIGHT," OR "MISSED-APPROACH POINT, RUNWAY AT (CLOCK POSITION)" OR "MISSED APPROACH POINT, APPROACH LIGHTS AT (CLOCK POSITION)"
FINAL (VFR) ³	500 FEET ABOVE FIELD ELEVATION	"500 ABOVE FIELD"
	100 FEET ABOVE FIELD ELEVATION	"100 ABOVE FIELD"
NOTES: 1. CHECK FOR APPEARANCE OF WARNING FLAGS AND GROSS INSTRUMENT DISCREPANCIES. CAPTAIN'S JUDGMENT ON EXCESSIVE ALTIMETER ERROR. 2. CARE MUST BE EXERCISED TO PRECLUDE CALLOUTS WHICH CAN INFLUENCE THE PILOT FLYING AND RESULT IN PREMATURE ABANDONMENT OF INSTRUMENT PROCEDURES. 3. IT IS RECOMMENDED THAT ALL AIRCRAFT UTILIZE ELECTRONIC/VISUAL SYSTEMS AS AN AID IN MAINTAINING GLIDESLOPE.		



UNUSUAL ATTITUDES

General

An unusual attitude is an aircraft attitude occurring inadvertently. It may result from one factor or a combination of several factors such as turbulence, distraction from cockpit duties, instrument failure, inattention, spatial disorientation, etc. In most instances, these attitudes are mild enough for the pilot to recover by re-establishing the proper attitude for the desired flight condition and resuming a normal cross-check.

Techniques of recovery should be compatible with the severity of the unusual attitude, the characteristics of the aircraft, and the altitude available for recovery.

The following aerodynamic principles and considerations are applicable to the recovery from unusual attitudes.

- The elimination of a bank in a dive aids in pitch control.
- The use of bank in a climb aids in pitch control.
- Power and speedbrakes, used properly, aid in airspeed control.

Recovery Procedures

Attitude Indicator(s) Operative

Normally, an unusual attitude is recognized in one of two ways: an unusual attitude picture on the attitude indicator or unusual performance on the performance instruments. Regardless of how the attitude is recognized, verify that an unusual attitude exists by comparing control and performance instrument indications prior to initiating recovery on the attitude indicator. This precludes entering an unusual attitude as a result of making control movements to correct for erroneous instrument indications. If there is any doubt as to proper attitude indicator operation, then recovery should be made using attitude indicator inoperative procedures:

- If diving, adjust power and/or speedbrakes as appropriate while rolling to a wings-level, upright attitude, and correct to level flight on the attitude indicator.
- If climbing, use power as required, and bank to the nearest horizon as necessary to assist in pitch control and to avoid negative G forces. As the aircraft symbol approaches the horizon bar, adjust pitch, bank, and power to complete the recovery and establish the desired aircraft attitude.

Attitude Indicator(s) Inoperative

With an inoperative attitude indicator, successful recovery from unusual attitudes depends greatly on early recognition of attitude indicator failure. For example, attitude indicator failure should be immediately suspected if control pressures are applied for a turn without corresponding attitude indicator changes. Another example is satisfactory performance instrument indications that contradict the picture on the attitude indicator.

If an unusual attitude is encountered with an inoperative attitude indicator, the following procedures are recommended:

- Check other attitude indicators, for proper operation and recover on the operative attitude indicator.

If no attitude indicator is operational:

- Determine whether the aircraft is in a climb or a dive by referring to the airspeed, altimeter, and vertical velocity indicators.
- If diving, roll to center the turn indicator and recover from the dive. Adjust power and/or speedbrakes as appropriate.

NOTE

Disregarding vertical attitudes, rolling away from the turn needle and then centering it results in an upright attitude.



- If climbing, use power as required. If the airspeed is low or decreasing rapidly, pitch control may be aided by maintaining a turn of approximately standard rate until reaching level flight.
- Upon reaching level flight, center the turn needle. Reversal of the altimeter and vertical velocity trends indicate passage of a level flight attitude.

TAKEOFF PROCEDURES AND FLIGHT PROFILES

Figures 18-4, 18-5, and 18-6 show accepted Citation 650 series aircraft takeoff profiles.

Prior to Takeoff

The minimum crew briefing prior to takeoff shall include the following:

1. A statement indicating the type of takeoff to be made (static, rolling, flaps 20°, flaps 7°, anti-ice on, etc.)
2. Takeoff thrust setting and airspeed callouts to be given. (In the checklist these items are reviewed prior to the crew briefing).
3. A statement of the intentions of the pilot in command in the event of an emergency/warning light prior to, or after, takeoff decision speed (V_1) to include emergency return/obstacles and MSA.
4. A statement of the intentions of the pilot in command with regard to heading and altitude to be flown for intended departure.
5. Radio and flight instrument setup
6. A request for "Any questions?" directed to all cockpit crewmembers

Sample Pretakeoff Briefing

1. "This will be a flaps [20 or 7] static [or rolling] takeoff." (Mention anti-ice if required.)
2. "Takeoff power and V-speeds have been briefed." (Review if desired.)
3. "Call out '60 knots,' '80 knots,' ' V_1 ,' 'rotate,' ' V_2 ,' and 'positive rate.'"
4. "Monitor all engine instruments and the annunciator panel during takeoff. At 60 knots call, 'adjust power.' At 80 knots call 'crosscheck both airspeed indicators.'"
5. "In the event of serious malfunction prior to V_1 , call 'abort'"
6. "If a malfunction occurs at or after V_1 , we will continue the takeoff. Advise me of the malfunction, and we will handle it as in in-flight emergency."
7. "For emergency return, plan to fly a _____ approach into _____ airport."
8. "Departure will be _____ heading to _____ feet."
9. "NAV 1 is tuned to _____ [navaid]. Tune NAV 2 to _____ [navaid]. My HSI has a course of _____ selected, the heading bug is on runway heading, and VOR [or ADF] is selected on my bearing pointer. Set the right HSI [as appropriate]."
10. "Highest obstacles on departure are _____."
11. "Any questions?"

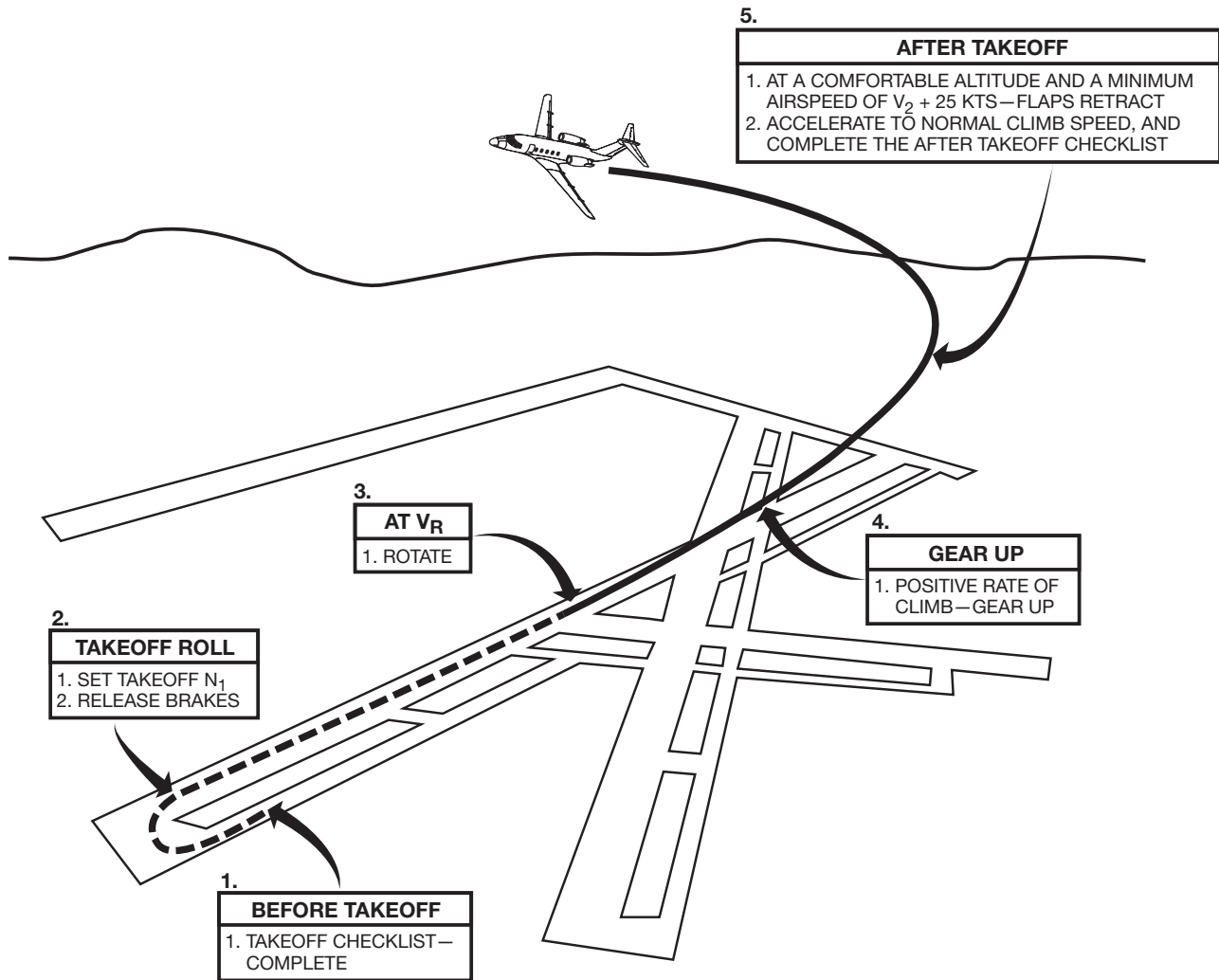


Figure 18-4. Takeoff—Normal

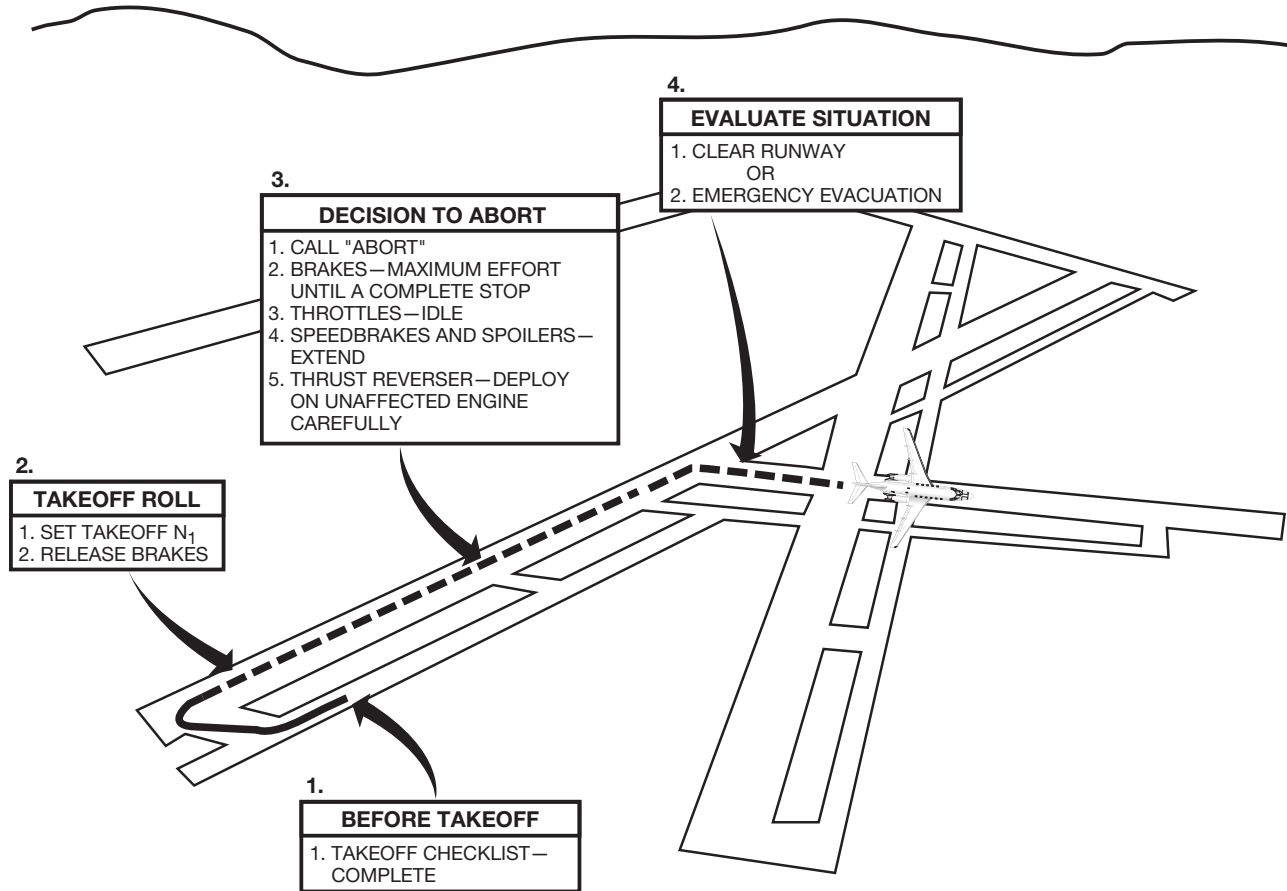


Figure 18-5. Engine Failure or Fire During Takeoff—Below V_1

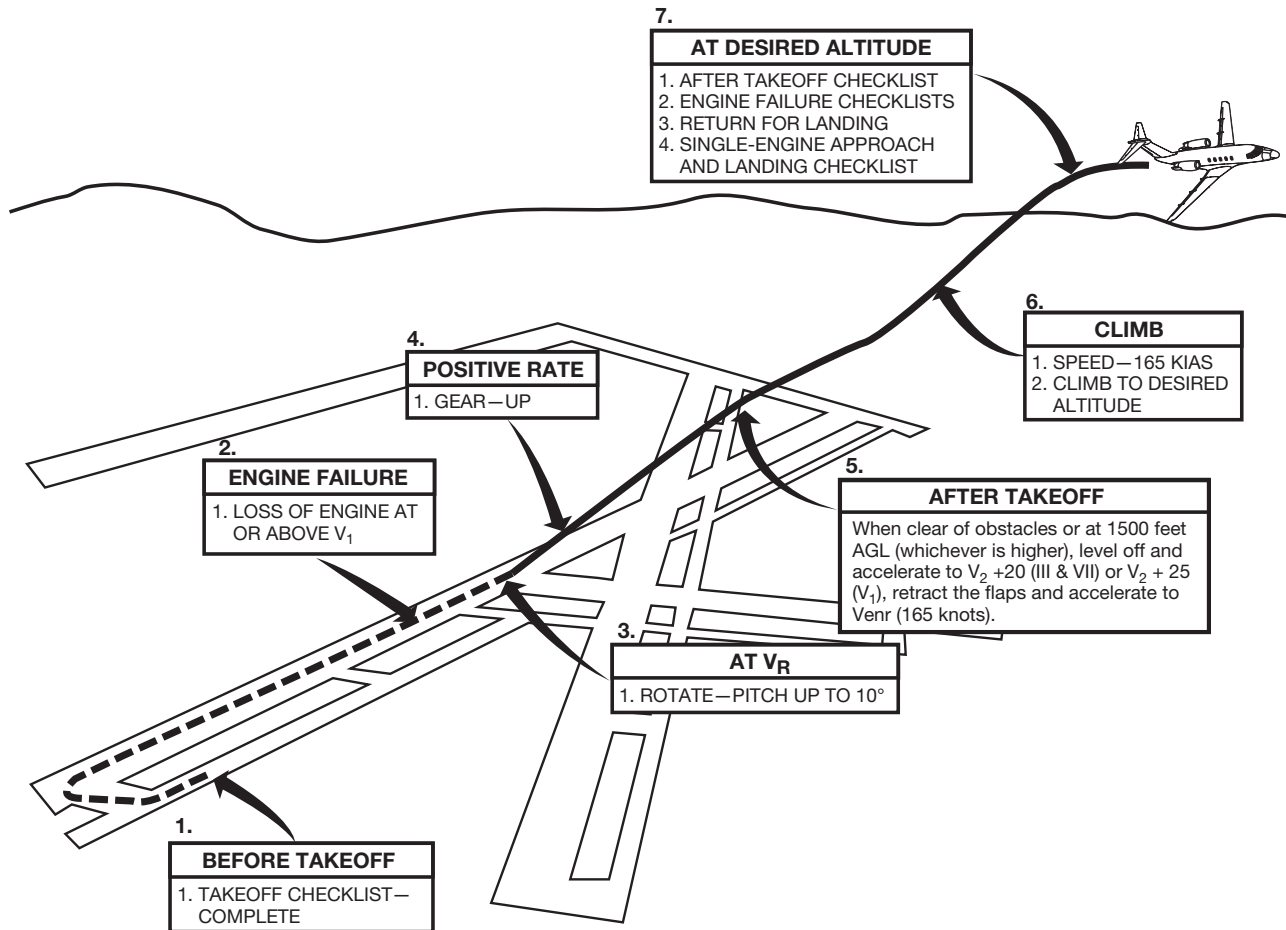


Figure 18-6. Takeoff with an Engine Failure at or After V_1



APPROACHES, LANDING PROCEDURES, AND FLIGHT PROFILES

Approach Briefing

The briefing should accomplish the following:

1. Identify the approach to be flown and the transition (Figure 18-7).
2. Assign the copilot to identify all intersections and the FAF.
3. Assign the NAV frequency and course SEL changes for the entire approach.
4. Assign the timing responsibility upon request.
5. Assign the copilot the standard callouts:
 - a. 1,000 feet before assigned altitudes
 - b. Localizer or course alive
 - c. 500 feet and 100 feet above published minimums
 - d. “Minimums, runway not in sight.” or “Minimums, runway in sight, cleared to land.”

Sample Approach Briefing

“We’ll fly the localizer back-course approach to runway 19L at Wichita. I want you to set 110.3 in NAV 1 with 013° in the course selector window. Also set 113.8 in NAV 2 with 081° course selected to identify KECHI. Set 332 on the ADF for PICHE. Identify all navigation aids. Start timing at KECHI, using two minutes thirty seconds for 120 knots groundspeed. After crossing KECHI, set the ILS frequency in NAV 2, and set your HSI to match mine. If we execute a missed approach, I’ll start a climb to 3,000 feet direct to IC LOM. We will observe all standard callouts.”

Landing Procedures and Flight Profiles

Figures 18-8 through 18-12 provide guidelines for several types of approaches.

NOTE

The term “ V_{REF} ” when it appears on the following diagrams, refers to V_{REF} per landing flap schedule (LFS). There are two normal landing flap schedules (settings)—full and 20°. Since V_{REF} is always used in this context to mean V_{REF} (per landing flap schedule), the parenthetical material has been omitted and the simpler expression “ V_{REF} ” used on the diagrams.

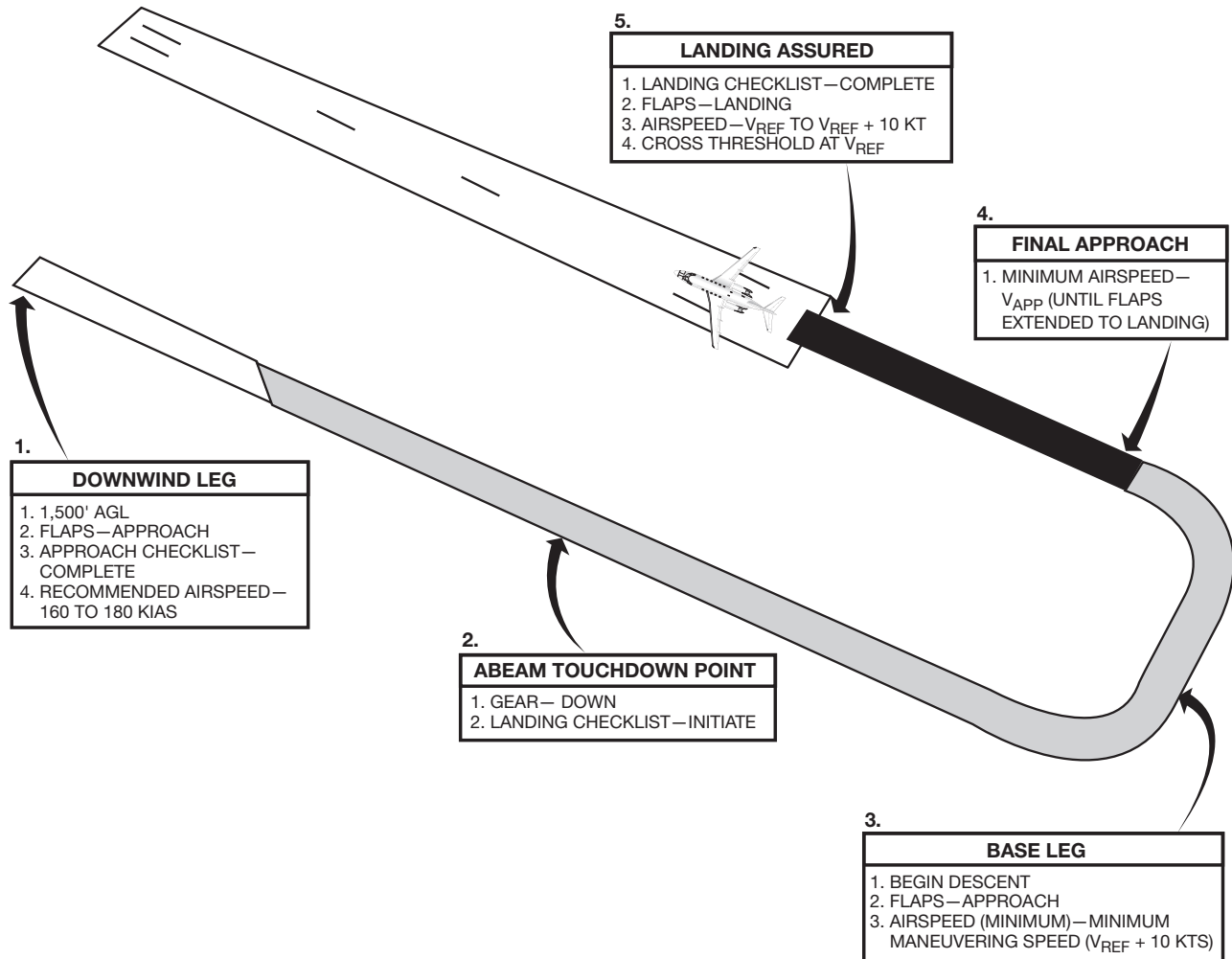


Figure 18-8. VFR Approach—Normal/Single Engine

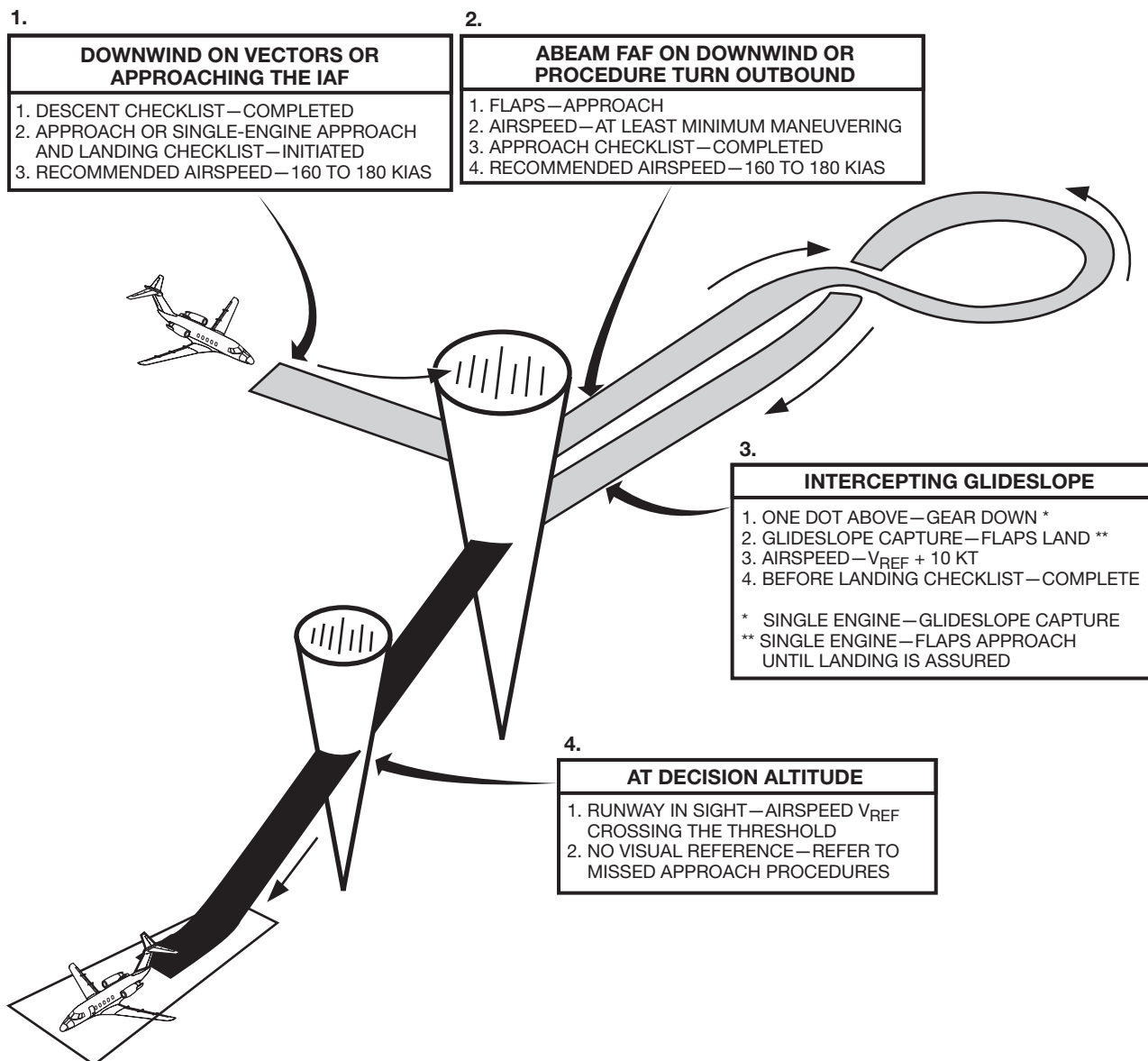
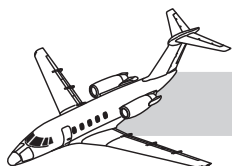


Figure 18-9. ILS Approach—Normal/Single Engine

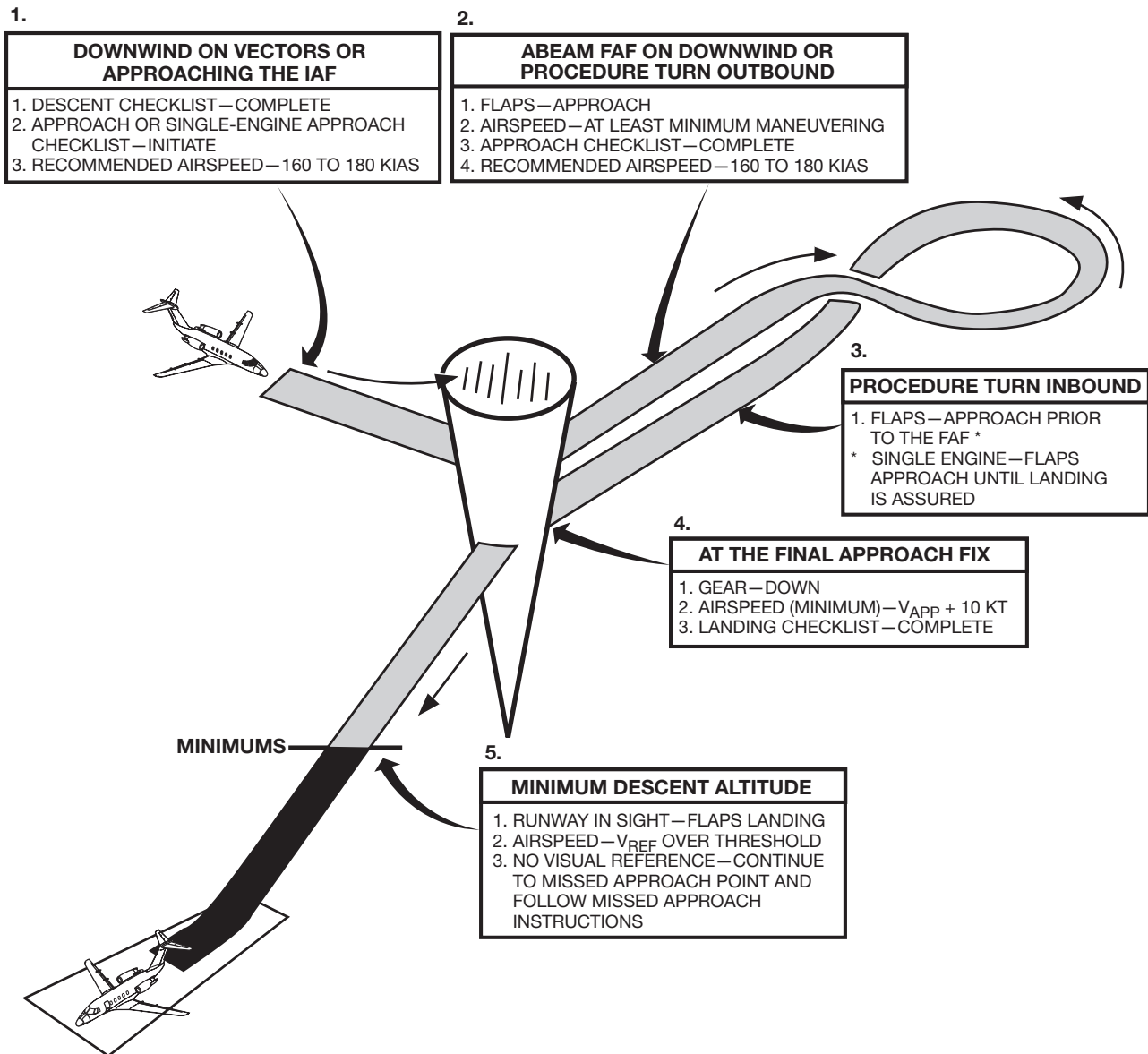


Figure 18-10. Nonprecision Approach—Normal/Single Engine

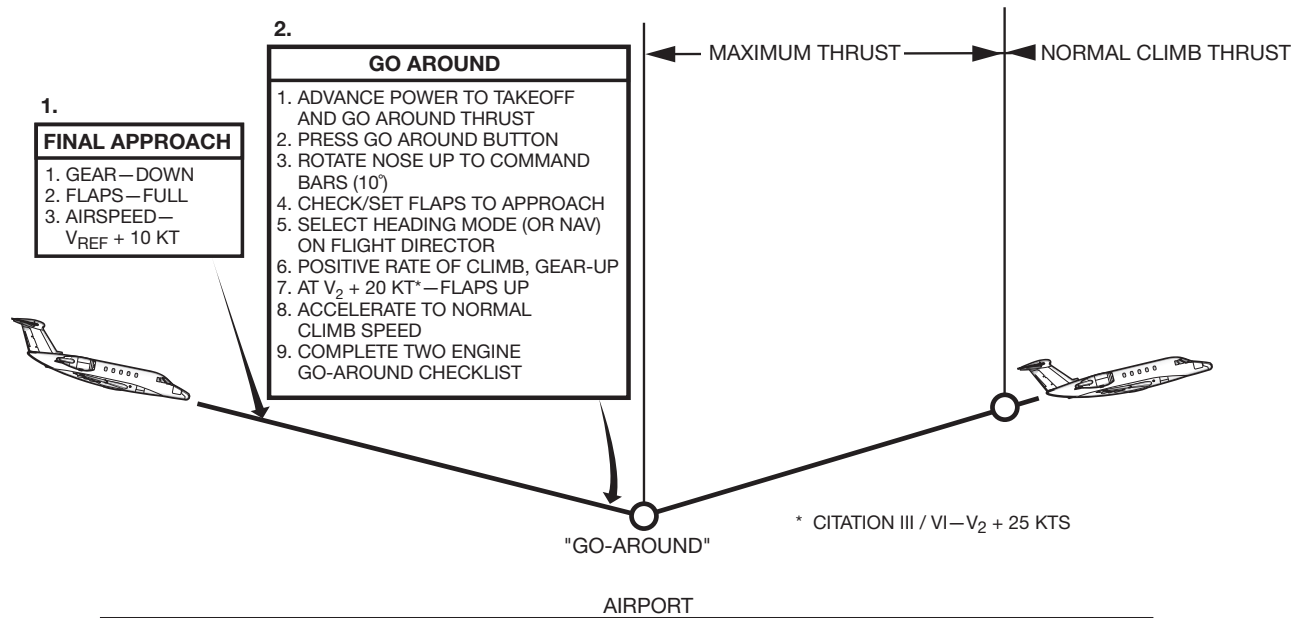


Figure 18-11. Missed Approach—Both Engines

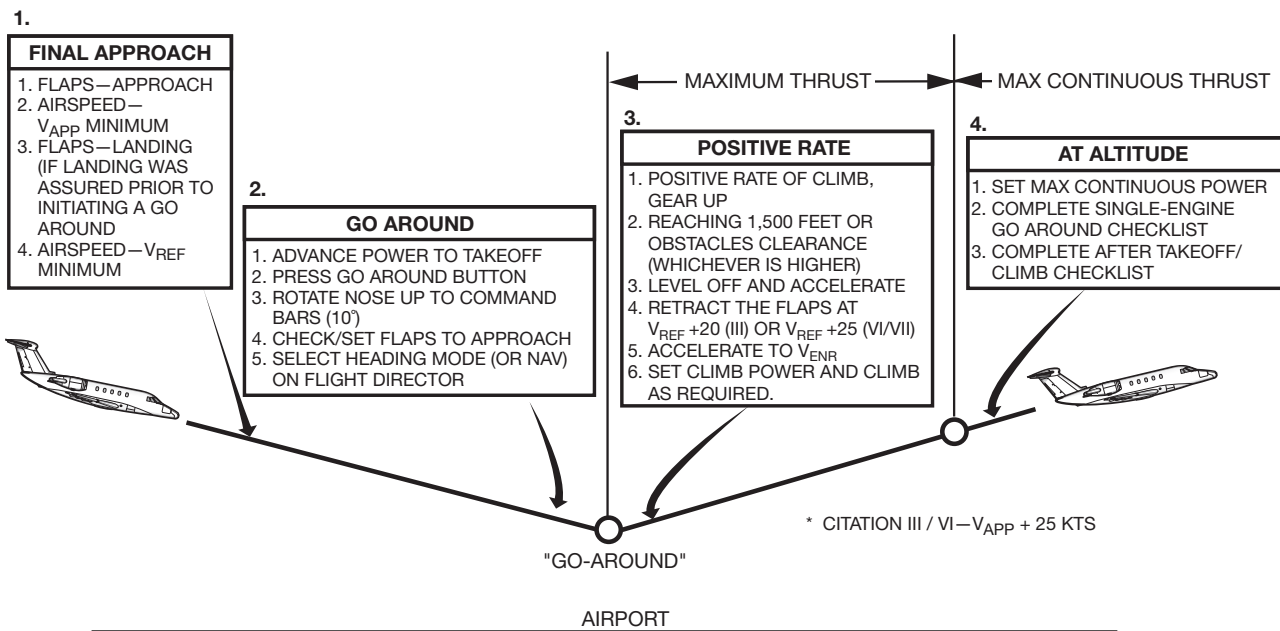


Figure 18-12. Missed Approach—One Engine



Landing Without Use of Thrust Reversers

Touchdown, preceded by a slight flare, should occur on the main wheels. Check thrust at idle, and extend the spoilers/speedbrakes while lowering the nosewheel. With the nosewheel on the runway, braking should be commenced according to runway length available to reduce brake wear. Apply smooth, gradually increasing pressure until a comfortable turn-off speed is reached. If maximum braking performance is desired, do not modulate brake pedals.

NOTE

Antiskid system dropout occurs at approximately 12 knots where braking reverts to straight power brake mode.

Landing With Thrust Reversers

Touchdown, preceded by a slight flare, should occur on the main wheels. Check thrust at idle, and extend the spoilers/speedbrakes and thrust reversers after lowering the nosewheel. Check thrust reverser DEPLOY and RUDDER BIAS annunciators are illuminated before adding reverse thrust. Apply the brakes and add reverse thrust after the nosewheel contacts the ground. There is a slight nose-up pitch associated with the deployment of the thrust reversers. Therefore, slight forward pressure should be used during the thrust reverser deployment, especially at high speeds such as a refused takeoff or a no-flap landing, to reduce this tendency. The nosewheel must be on the ground before adding reverser power to eliminate the possibility of foreign object damage and to improve directional control.

CAUTION

Do not attempt to restow reversers and takeoff once reversers have started to deploy. Throttle linkage damage may occur, resulting in loss of power or flameout.

Once the thrust reversers are deployed, move the thrust reverser levers aft to obtain maximum reverse thrust. Stops have been installed on the thrust reverser levers to limit the maximum thrust available so as not to exceed the takeoff thrust rating. This allows the pilot to keep his attention on the landing rollout instead of diverting his attention to the reverse power settings, except in abnormally cold ambient temperature conditions. The thrust reverser levers should be placed in idle reverse by 65 knots to prevent foreign object damage to the engine.

NOTE

Use of thrust reversers is not permitted during touch-and-go landings.

NOTE

Due to possible FOD to the engine during taxi, keep the use of the thrust reversers to a minimum.

Adjustments to Landing Distance

- Antiskid Inoperative—Refer to Antiskid Inoperative Checklist
- 0° Flaps—Refer to the Flaps Up or 7° Checklist
- 7° Flaps—Refer to the Flaps Up or 7° Checklist
- Wet Runway*—Refer to advisory information in the *AFM*
- Icy Runway*—Refer to advisory information in the *AFM*
- * For thrust reverser utilization, data from the *Operating Manual* Section VII should be consulted.



Crosswind Landing

Method No. 1:

- The aircraft can be flown down final approach with runway centerline alignment maintained with normal drift correction. Approaching the threshold, lower the upwind wing to maintain no drift, and apply opposite rudder to maintain alignment with runway centerline. Fly the aircraft onto the runway—do not allow drift to develop.

Method No. 2:

- The crab or wings-level method may be continued until just before touchdown. Then, with wings level, apply rudder pressure to align the aircraft with the runway centerline at the moment of touchdown. Fly the aircraft onto the runway—do not allow drift to develop.

Circling Approaches

A circling approach may follow any authorized instrument approach. Although the Citation 650 series aircraft are in approach category C, category D minimums may be used during the circling approach due to the higher maneuvering airspeeds. A normal instrument approach is flown until visual contact with

the runway environment is made at the MDA published for the particular circling approach. With the airport in sight, the approach becomes a visual reference approach with a continued cross-check of the flight instruments. Since it is primarily a visual approach at this point, configuration and speeds are the same as for a normal visual approach.

Leaving the final approach fix, maintain minimum maneuvering speed with the flaps at approach flap schedule and the landing gear down. Reduce the power to provide a 1,000 foot-per-minute rate of descent. When approaching MDA, power should be added to maintain airspeed while leveling off, thereby reducing the rate of descent and ensuring that the aircraft does not go below MDA. There are many recommended circling procedures once the airport is in sight. Any procedure is acceptable, provided the following criteria are met:

- The airport environment is always in sight.
- A safe and controllable airspeed is maintained.
- MDA is maintained until the aircraft is in position to perform a normal descent to a landing on the landing runway without excessive maneuvering.



VISUAL APPROACH AND LANDING WITH FLAPS INOPERATIVE (UP OR 7°)

When planning a flaps-inoperative approach and landing (Figure 18-13), the landing weight of the aircraft must be considered. An attempt should be made to reduce this weight if possible, especially if runway length is marginal due to the higher approach and landing speeds required for a no-flap configuration. Compute adjusted V_{REF} for 0° flaps or 7° flaps as appropriate from the checklist. Set the airspeed indexer (airspeed bug) on the new adjusted V_{REF} speed. Fly the final approach at a maximum speed of the adjusted $V_{REF} + 10$ knots, and reduce to the adjusted V_{REF} prior to crossing the threshold.

To preclude excessive float during landing, allow the aircraft to touch down in a slightly flatter attitude than on a normal landing.

APPROACH TO STALL AND FLIGHT PROFILES

Prior to any planned approaches to stall (Figures 18-14 through 18-16), clear the area visually. All recoveries are made with power and a minimum loss of altitude.

At least one approach to a stall shall be made in other than straight flight, between 15–30° bank.

Stall warning is provided by stick shakers attached to the control columns. They are activated at an angle-of-attack indication of 0.82 ± 0.02 . Stall warning is also achieved aerodynamically, aided by stall strips on the inboard section of each wing. These strips disrupt the airflow over the horizontal stabilizer, resulting in a pre-stall buffet. Stall recovery should be initiated at the onset of either indication.

Prior to stalls:

1. Ignition..... ON
2. Compute climb power setting for altitude and temperature.
3. Compute V_{REF} for gross weight.
4. Engine Synchronizer OFF
5. Yaw Damper..... OFF

Limitation—No intentional stalls are permitted above 20,000 feet MSL.

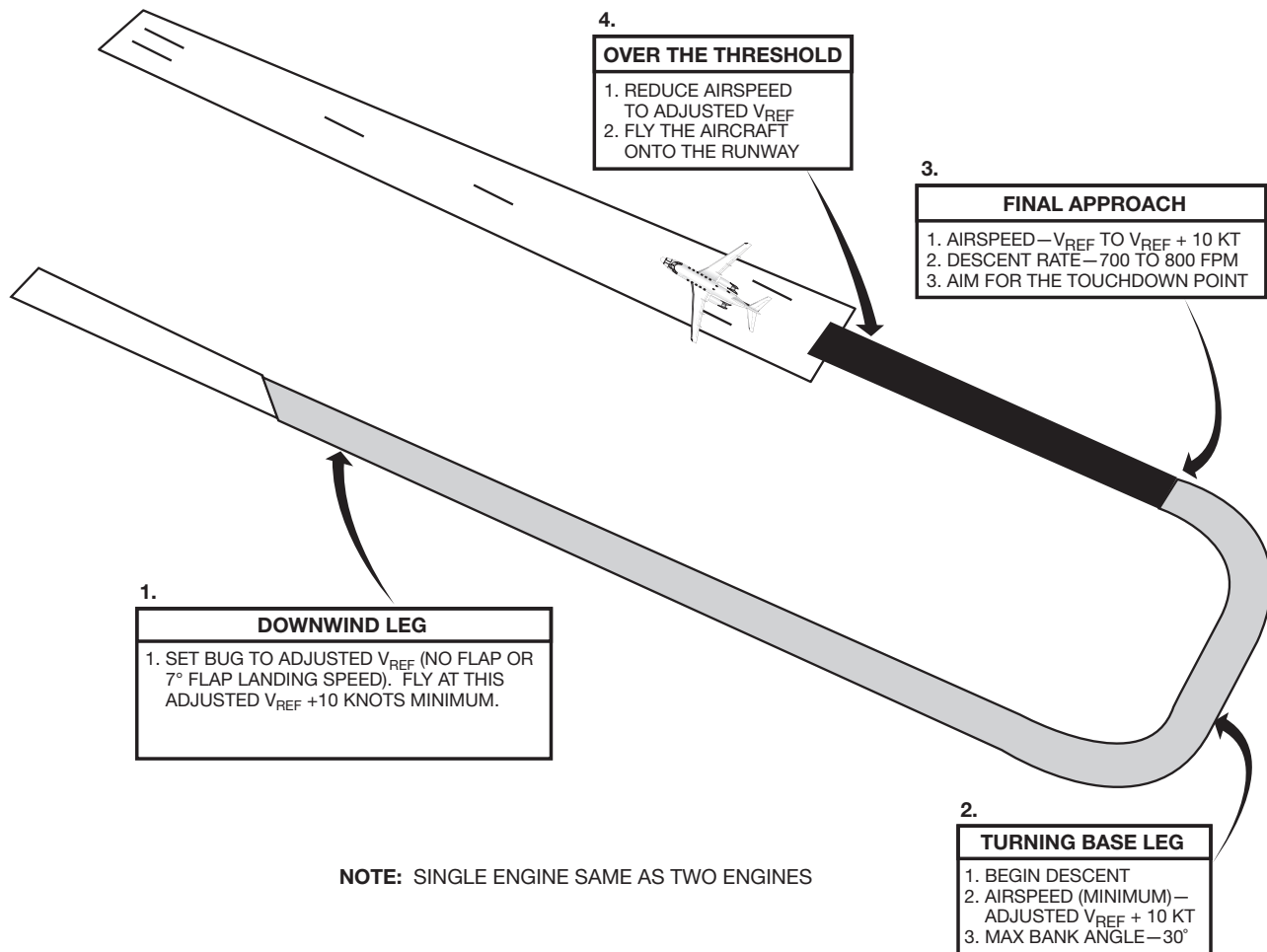


Figure 18-13. Visual Approach and Landing with Flaps Inoperative



CITATION 650 SERIES PILOT TRAINING MANUAL

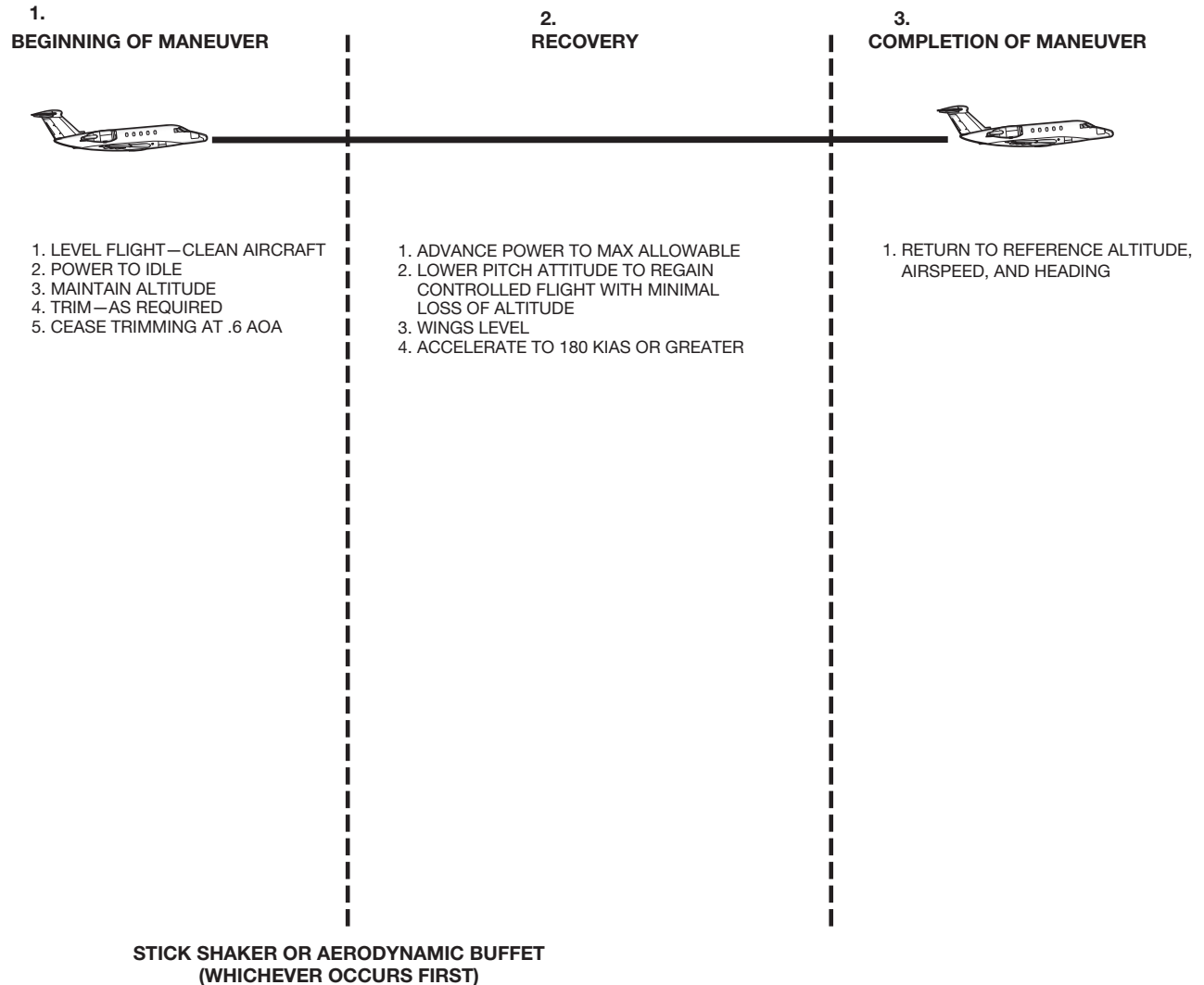


Figure 18-14. Approach to Stall—Enroute Configuration

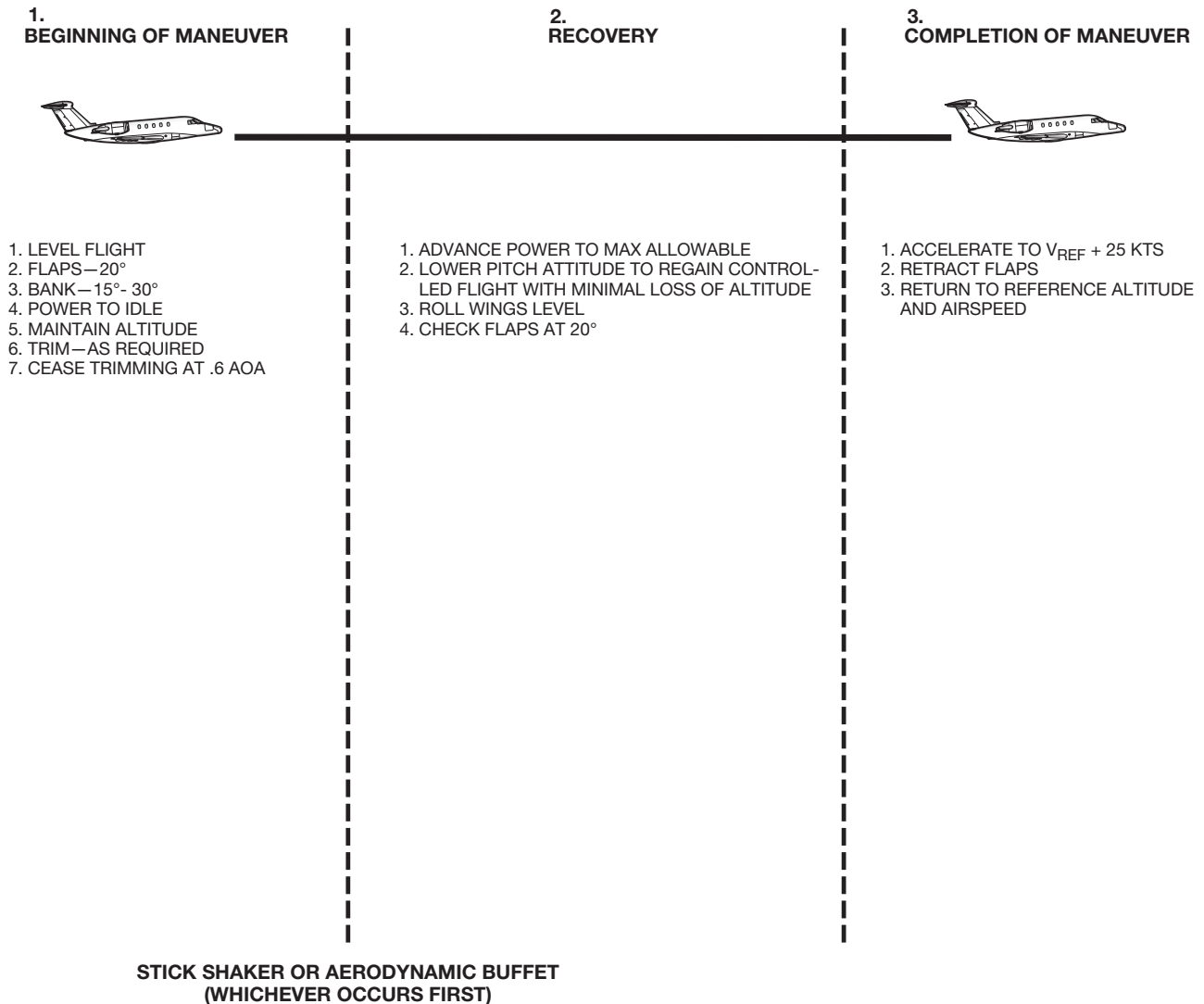


Figure 18-15. Approach to Stall—Takeoff Configuration



1. BEGINNING OF MANEUVER



1. LEVEL FLIGHT
2. FLAPS—20°
3. GEAR—DOWN
3. FLAPS—FULL
4. POWER TO 60% N_1
5. MAINTAIN ALTITUDE
6. TRIM—AS REQUIRED
7. CEASE TRIMMING AT .6 AOA
8. INITIATE RECOVERY AT FIRST INDICATION OF APPROACH TO STALL

2. RECOVERY

1. ADVANCE POWER TO MAX ALLOWABLE
2. LOWER PITCH ATTITUDE TO REGAIN CONTROLLED FLIGHT WITH MINIMAL LOSS OF ALTITUDE
3. WINGS LEVEL
4. FLAPS TO 20°
5. POSITIVE RATE OF CLIMB—GEAR UP

3. COMPLETION OF MANEUVER



1. ACCELERATE TO $V_{REF} + 25$ KTS
2. RETRACT FLAPS
3. RETURN TO REFERENCE ALTITUDE, AIRSPEED, AND HEADING

STICK SHAKER OR AERODYNAMIC BUFFET
(WHICHEVER OCCURS FIRST)

Figure 18-16. Approach to Stall—Landing Configuration

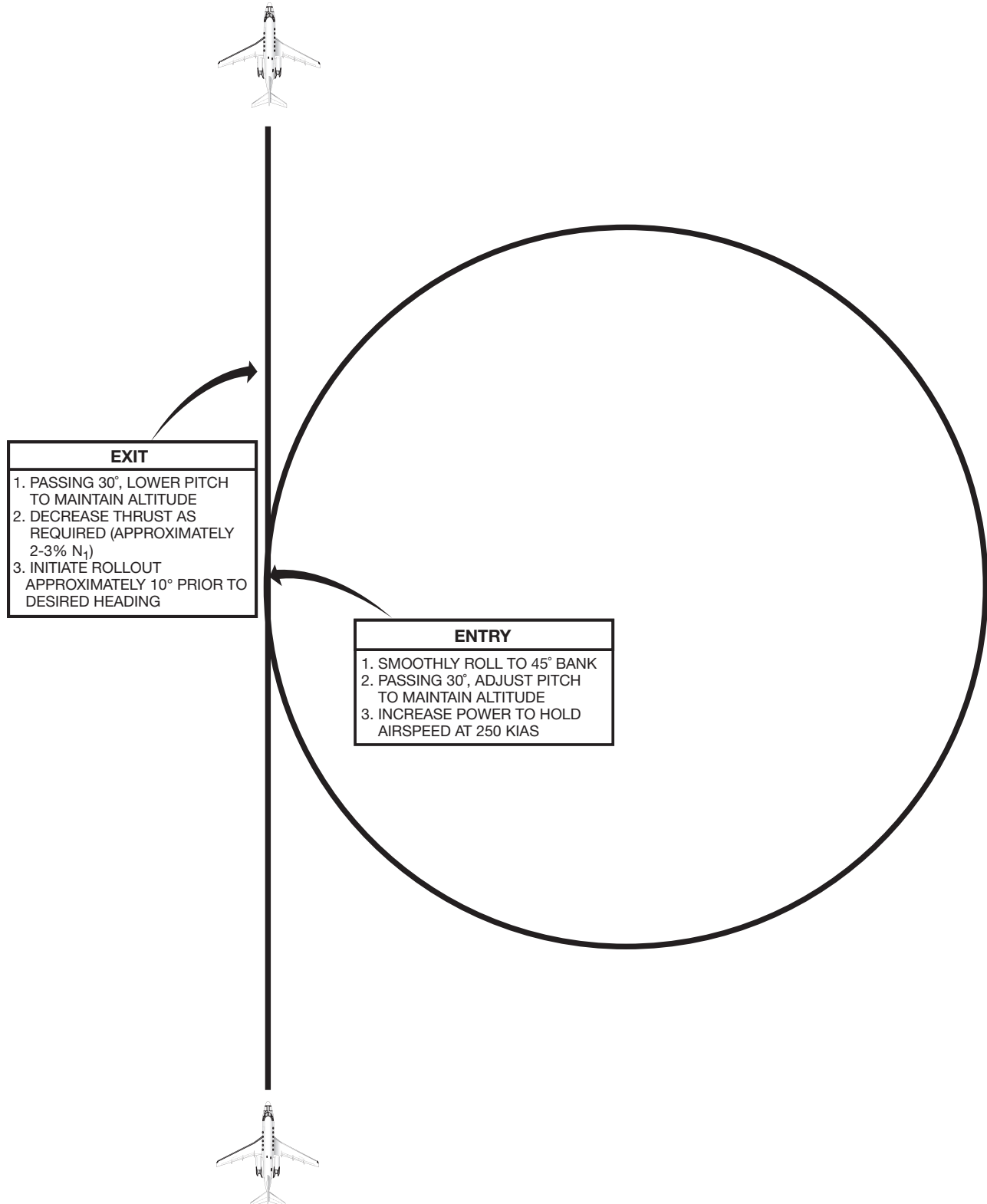


Figure 18-17. Steep Turns



EMERGENCY DESCENT AND FLIGHT PROFILE

1. Start maneuver (Figure 18-18) at an altitude of 35,000 to 45,000 feet.
2. To establish a manual emergency descent, the throttles must be brought to idle, the spoilers deployed, and an initial pitch attitude of approximately 20° nose down should be used. Lateral control is reduced with the spoilers extended.

3. Call out periodic altitude checks during descent.
4. Copilot calls 2,000 feet above level-off altitude; start level-off 1,000 feet above altitude, and retract spoilers/speedbrakes.

NOTE

Do not let airspeed decay below 150 KIAS with spoilers/speedbrakes extended.

1.

DECISION TO MAKE EMERGENCY DESCENT

1. CREW OXYGEN MASKS—DON AND 100% *
2. PASSENGER OXYGEN—ON
3. OXYGEN MICROPHONE SWITCHES—MIC OXY MASK

* NOTE:

IF SMOKE IS PRESENT IN THE COCKPIT, SELECT THE OXYGEN MASKS TO EMER AND LEAVE MASKS ON UNTIL SMOKE HAS BEEN EVACUATED.

2.

INITIATE DESCENT

1. THROTTLES—IDLE
2. SPEED BRAKES AND SPOILERS—DEPLOY
3. ALLOW NOSE TO LOWER TO 20° AVOIDING NEGATIVE Gs
4. SPEED— V_{MO}/M_{MO} (USE REDUCED SPEED IF THERE IS STRUCTURAL DAMAGE)
5. ENSURE PASSENGER OXYGEN ON (AS REQUIRED)

3.

DURING DESCENT

1. ATC—NOTIFY
2. ATC TRANSPONDER—7700 (IF NECESSARY)
3. ALTIMETER SETTING—REQUEST
4. DETERMINE MINIMUM SAFE ALTITUDE
5. PRESSURIZATION—RESET IF ABLE

4.

APPROACHING DESIRED ALTITUDE

1. LEVEL OFF—INITIATE 1,000' PRIOR TO DESIRED ALTITUDE
2. SPEED BRAKES—RETRACT
3. SPOILERS—RETRACT

Figure 18-18. Emergency Descent



SPECIAL PROCEDURES

ANTI-ICING ADDITIVES

Although the use of anti-ice additive is not necessary for icing conditions on aircraft equipped with fuel heaters, it is recommended that it still be used in nontreated fuel to control bacteria and fungi. The approved anti-ice additive is biocidal, which means it controls microorganisms such as bacteria and fungi that can rapidly multiply and cause serious corrosion in tanks and may block filters, screens, and fuel metering equipment.

Procedure for Adding Fuel Additive

When the aircraft is being refueled through the overwing ports, use the following procedure to blend anti-icing additive to nontreated fuel:

1. Attach MIL-I-27686E inhibitor to the refuel nozzle, making sure blender tube discharges in the refueling stream.
2. Start refueling (minimum 30 gallons per minute, maximum 60 gallons per minute) while simultaneously fully depressing and slipping ring over trigger of blender (rate of less than 30 gallons per minute may be used when “topping off”).

WARNING

Anti-icing additives containing ethylene glycol monomethyl ether (EGME) are harmful if inhaled, swallowed, or absorbed through the skin, and cause eye irritation. Also, it is combustible. Before using this material, refer to all safety information on the container.

CAUTION

Assure that the additive is directed into the flowing fuel stream and that the additive flow is started after the fuel flow starts and is stopped before fuel flow stops. Do not allow concentrated additive to contact coated interior of fuel tank or aircraft painted surface. Use not less than 20 fluid ounces of additive per 260 gallons of fuel or more than 20 fluid ounces of additive per 104 gallons of fuel.

Procedure for Checking Fuel Additive

Prolonged storage of the aircraft results in a water buildup in the fuel which leaches out the additive. An indication of this is an excessive amount of water accumulated in the fuel tank sumps. The concentration of additive can be checked using a CJMD 128-002 anti-icing additive concentration test kit available from Cessna Aircraft Company, Citation Marketing Division. It is imperative that the instructions for the test kit be followed explicitly when checking the additive concentration. The minimum additive concentration shall be 0.06% by volume, and maximum concentration shall be 0.15% by volume. Fuel, when added to the tank, should have a minimum concentration of 0.06% by volume.

SHORT-FIELD OPERATION

For takeoff, taxi into position as close to the approach end as possible, and apply takeoff thrust while holding the brakes. *AFM* takeoff field length data assumes a static run-up and use of all available runway. When specified thrust is set, release the brakes. Rotate smoothly precisely at V_R , as a delay results in degradation of takeoff climb performance. Retract the gear when positively climbing, and climb at V_2 with 20° flaps until clear of any obstacles. Retract the flaps at V_2+25 KIAS when clear of obstacles.



Landing field length data in the *AFM* assumes a threshold crossing speed of V_{REF} at an altitude of 50 feet with thrust reduced to idle at that point, and a 3° approach was maintained to touchdown. In practice, it is suggested that for minimum field operations the threshold be crossed at a comfortable obstacle clearance altitude, allowing some deceleration to take place approaching the runway. Touchdown should occur with maximum available runway remaining at minimum safe speed.

The energy to be dissipated during rollout is directly related to aircraft weight and velocity at touchdown. Although weight is normally dictated by cabin loading and reserves required, flight planning into short fields should include avoiding carrying excessive weight in stored fuel. This consideration offers the side benefit of improved enroute performance. Velocity is something that can be controlled in nearly every case. Precise speed control is important in the short field environment. A 1% increase in landing speed requires approximately 2% more rollout distance.

In general, short field landings are accomplished the same as normal landings except for heavier braking and closer attention to touchdown point and speed. A stabilized approach at V_{REF} provides the best possible starting point because any corrections necessary are small. Establish a glide angle that safely clears any obstacles and result in touchdown as comfortably close to the approach end as feasible. Avoid a very flat approach, as it generally results in excessive power being required to touch down and the vertical gust protection margin is reduced. At approximately 50 feet AGL, power reduction is normally begun to cross the threshold at a speed not in excess of V_{REF} . Check that the throttles are at idle, and avoid an excessive flare that may cause the aircraft to float. Deceleration takes place much more rapidly on the runway than airborne.

If thrust reversers are not used, extend the spoilers while lowering the nose, and commence braking with steadily increasing pressure. Once braking has begun, back pressure on the yoke creates elevator drag while slightly

increasing weight on the gear, provided the nosewheel is not lifted off the runway.

For landings utilizing thrust reversers, after touchdown on the mains, lower the nose, apply wheel brakes, extend spoilers/speedbrakes, and deploy the thrust reversers. Slight forward pressure on the yoke should be applied during reverser deployment. Check illumination of the ARM, UNLOCK, and DEPLOY lights and the RUDDER BIAS annunciator. Once the thrust reversers are deployed, apply maximum reverse thrust power. Once braking has begun and maximum reverse power is reached, back pressure on the yoke provides additional weight on the main gear, provided the nose is not raised. At 65 KIAS return the thrust reverser levers to the idle reverse detent position. Leave the thrust reversers deployed for aerodynamic drag and idle reverse thrust.

ADVERSE FIELD CONDITIONS

All *AFM* field length data assumes a dry, hard-surface runway except where otherwise noted. Precipitation-covered runway conditions degrade braking effectiveness and require significantly greater actual takeoff abort and landing field lengths.

Considerations for landing on a precipitation-covered runway are similar to those for short-field operations where velocity and speed are minimized and maximum rollout distance is made available. Runway composition, condition and construction, the amount of precipitation, and the depth of main landing gear tire tread remaining affect the magnitude of braking degradation, so it is impossible to apply a fixed factor to cover all conditions. Refer to the FAA-approved *AFM* Section VII, "Advisory Information", for data that permits estimation of the minimum runway required under various precipitation-covered runway conditions. Again, maximizing rollout runway available and touching down at minimum safe speed provides the greatest possible margin.

Use of the thrust reversers on precipitation-covered runways is the same as that for a landing on a normal or dry runway. Cockpit



visibility is not hampered by blowing rain, snow, or ice thrown forward by the thrust reverser. Single-engine reversing during cross-wind landings on precipitation-covered runways should be used with discretion.

With precipitation cover on the runway, braking should be very judicious. If runway length permits, delay braking slightly until some aerodynamic deceleration has taken place. Under normal braking conditions, the antiskid system is very effective in preventing skids and producing minimum stopping distance. However, on a precipitation-covered runway, the phenomena of hydroplaning may greatly reduce the antiskid effectiveness due to the possibility of the aircraft wheels not rotating up to a speed equal to the aircraft groundspeed. With 168-psi main gear tire inflation the minimum dynamic hydroplaning initiating groundspeed may occur at speeds above approximately 91 knots. Since groundspeed is the critical factor, landing on precipitation-covered runways with any tailwind component should be avoided. Good tread depth tends to relieve hydrodynamic pressure under the tire on wet runways, and inflation is important because a low tire pressure lowers the minimum hydroplaning speed. Anticipated operation on precipitation-covered runways dictates close monitoring of tire condition and pressure.

After landing on ice or slush, a complete check of the aircraft, including overboard vents and control surfaces, should be conducted.

ENGINE ANTI-ICE

The importance of proper system use cannot be overemphasized, as serious engine damage can result from ice ingestion. Engine anti-ice function is preventative in nature, and flight into visible moisture with a temperature of $+10^{\circ}\text{C}$ TAT/RAT or less should be anticipated so that the system is operational when icing conditions are encountered. Turning the engine anti-ice on after ice has accumulated could result in ice from the inlet and inboard wing leading edge being freed and ingested by the engine. For this reason, wing anti-ice should also be used when engine anti-ice is required.

In descent, the engine anti-ice should be turned on well before entering an icing environment to ensure that sufficient time is available for all system parameters to be met.

Ice formation observed on the wing is not a valid indication of possible engine icing. The ENGINE ANTI-ICE switches should be ON and operating any time there is visible moisture with a temperature $+10^{\circ}\text{C}$ TAT/RAT or less.

PASSENGER COMFORT

When parked during daylight in hot weather, it is suggested that the cabin window shades be closed to reduce solar heat transfer. An optional exterior windshield cover performs the same function for the cockpit and is very effective. To circulate cool air in the interior, either engine or the optional onboard auxiliary power unit may be started, the isolation valve opened, and the CKPT/CAB PAC switch positioned to HIGH. Temperature is then controlled with the CKPT/CAB TEMP SEL hot/cold switch. Closing unused overhead outlets directs airflow to the occupied seats. Increasing air circulation in the cockpit is available by turning on the W/S DEFOG fan. Operating the right engine above idle rpm increases airflow and environmental control unit efficiency. Operation of an onboard auxiliary power unit (if installed) runs the environmental control unit without the need to start the aircraft engines.

Additional heat to the cockpit can be provided by turning on the auxiliary electric heater (Citation III and VI aircraft only).

The abbreviated checklist is designed to enable the crew to perform all prestart functions in advance. This permits items such as the warning test to be completed before passenger boarding and accelerates the ramp departure without compromising safety or thoroughness.

Leaving the chocks, brake checks can be done lightly and smoothly. If heavy braking is required on landing roll, using up elevator to create drag also counters the nosedown pitching moment so that deceleration feel in the cabin is less abrupt. Do not apply excessive back pressure, however, which could result in



raising the aircraft nose and/or increasing the possibility of a blown tire.

The pressurization system procedures outlined in this chapter may at first appear complex, but thorough understanding of the controller and indicators coupled with minimal practical experience greatly simplifies operation. Optimum system performance in terms of passenger comfort is best achieved by slow, smooth selection of altitudes and rates, and reducing the variables when setting the controller by not making power changes simultaneously. Lightweight takeoffs and/or rolling takeoffs in cold ambient conditions may cause pressurization oscillations (bumps) as the aircraft lifts off, because the cabin did not have sufficient time to prepressurize. Cabin pressure oscillations can be prevented by allowing the prepressurization to begin before the brakes are released. This procedure allows the cabin to correctly prepressurize prior to lift-off and provides a smooth transition to pressurized flight.

Although it is not mandatory, use of the yaw damper is recommended when hand-flying the aircraft. It reduces pilot rudder input required, and the aircraft rides better in rough air. The yaw damper must be off for takeoff and landing.

Power management has an impact on cabin comfort, and changes should be made smoothly and symmetrically. An approximate estimate of synchronization can be made by observing the RPM gauges and exact adjustments made with the engine synchronizer. Although the higher-pitched turbine sound is generally more noticeable in the cockpit, the lower-pitched, fan-out-of-synchronization condition is usually more prevalent in the area of the rear seats.

Good crew coordination and smooth operation of the controls and systems serve the best interests of safety, economy, and passenger comfort.

COMMUNICATION

During ground operation, radio transmissions can be blocked by surrounding terrain or structures. This may be possible to overcome by

using the other COMM, because of aircraft antenna location. The COMM 1 antenna is on the underside of the fuselage, and the COMM 2 antenna is on the upper side. Flying through dry precipitation, it is possible for static electricity buildup to cause the very-high-frequency communication systems to automatically squelch to a point where reception range is greatly reduced. Disabling the squelch by depressing the SQ knob causes background static in the speaker or headset, but normal reception range is restored. If the headset microphone fails to function properly, check that the appropriate instrument subpanel switch is in the MIC HEAD SET position.

COLD-WEATHER OPERATION

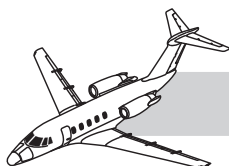
If the aircraft has been cold-soaked below -18°C with dual-battery installation or below 0°C with single-battery installation, a ground power unit or the onboard auxiliary power unit and /or preheat should be used for starting. If a start is attempted and the starter does not motor to 10% N_2 minimum, terminate the sequence. Advancing the throttle to idle below 10% N_2 can be damaging to the engine and battery.

Engine preheating is best accomplished by installing the engine covers and directing hot air through the oil filler access door. A warm battery provides some benefit, and the heater hose can be placed in the tailcone with the door propped as far closed as possible to minimize heat loss. With sufficient hose length, the cabin and cockpit area can be warmed through the pilot side window.

Operating in extremely cold temperatures reduces fuel/water solubility and super cools any water particles in the fuel, increasing the possibility of fuel system icing. The tank and fuel filter drains under each wing should be drained frequently and thoroughly. It is possible for water to settle in the sump and freeze, blocking the drain, in which case heat should be applied until fuel flows freely. Maintain heat after flow begins to ensure that all particles have melted, and collect the drainage in a clear, clean container to inspect for water globules.



The information normally contained in this chapter can be found
in the *Operating Manual* and *Airplane Flight Manual*.



CHAPTER 20

FLIGHT PLANNING AND PERFORMANCE

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General	20-2
Formulas	20-2

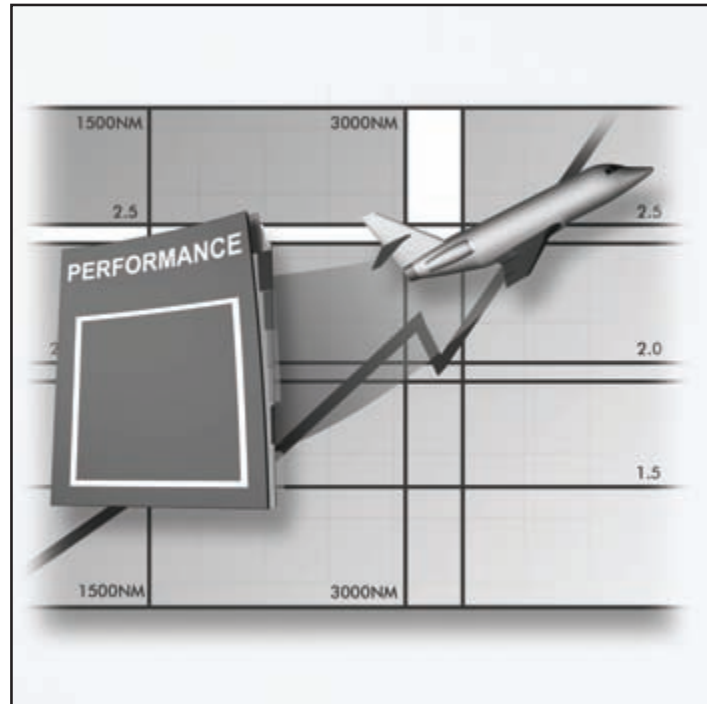


ILLUSTRATIONS

Figure	Title	Page
20-1	Calculation of Takeoff Performance.....	20-3
20-2	Calculation of Landing Performance	20-4



CHAPTER 20 FLIGHT PLANNING AND PERFORMANCE



INTRODUCTION

Performance is calculated using a combination of charts and tables in the *Aircraft Flight Manual* and the *Aircraft Performance Manual*. The takeoff and landing performance data is found in Section IV—“Performance” and Section VII “Advisory” of the *AFM*. The climb, cruise, and descent performance data is found in the *Performance Manual*.



GENERAL

This aircraft is certified to Part 25 standards. Keep in mind that the FAR Part 25 performance requirements do not meet the minimum requirements (3.3% or 200 ft/nm) of the FAA “IFR Takeoff Flight Path”.

The maximum takeoff weight—pounds permitted by climb requirements chart only guarantees second segment climb performance, not any of the other segments.

The following are the minimum climb gradients as specified by FAR Part 25:

- 1st segment 0% gross
- 2nd segment..... 1.6% net
- 3rd segment N/A
- Final segment 1.2% gross

NOTE

The gross climb gradient reduced by a required factor and used for calculation of take-off flight path.

A simplified block diagram of the calculation of takeoff performance is illustrated in Figure 20-1. A simplified block diagram of the calculation of landing performance is illustrated in Figure 20-2.

FORMULAS

Runway Slope =

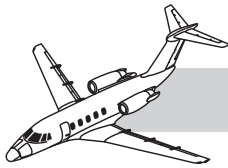
$$\frac{\text{Change in Elevation Between Ends of the Runway (Rise)}}{\text{Runway Length (Run)}} \times 100$$

Gradient (in %) =

$$\frac{\text{Feet per NM} \times 100}{6076}$$

Climb rate (feet per minute) =

$$\text{Groundspeed} \times \text{Gradient}$$



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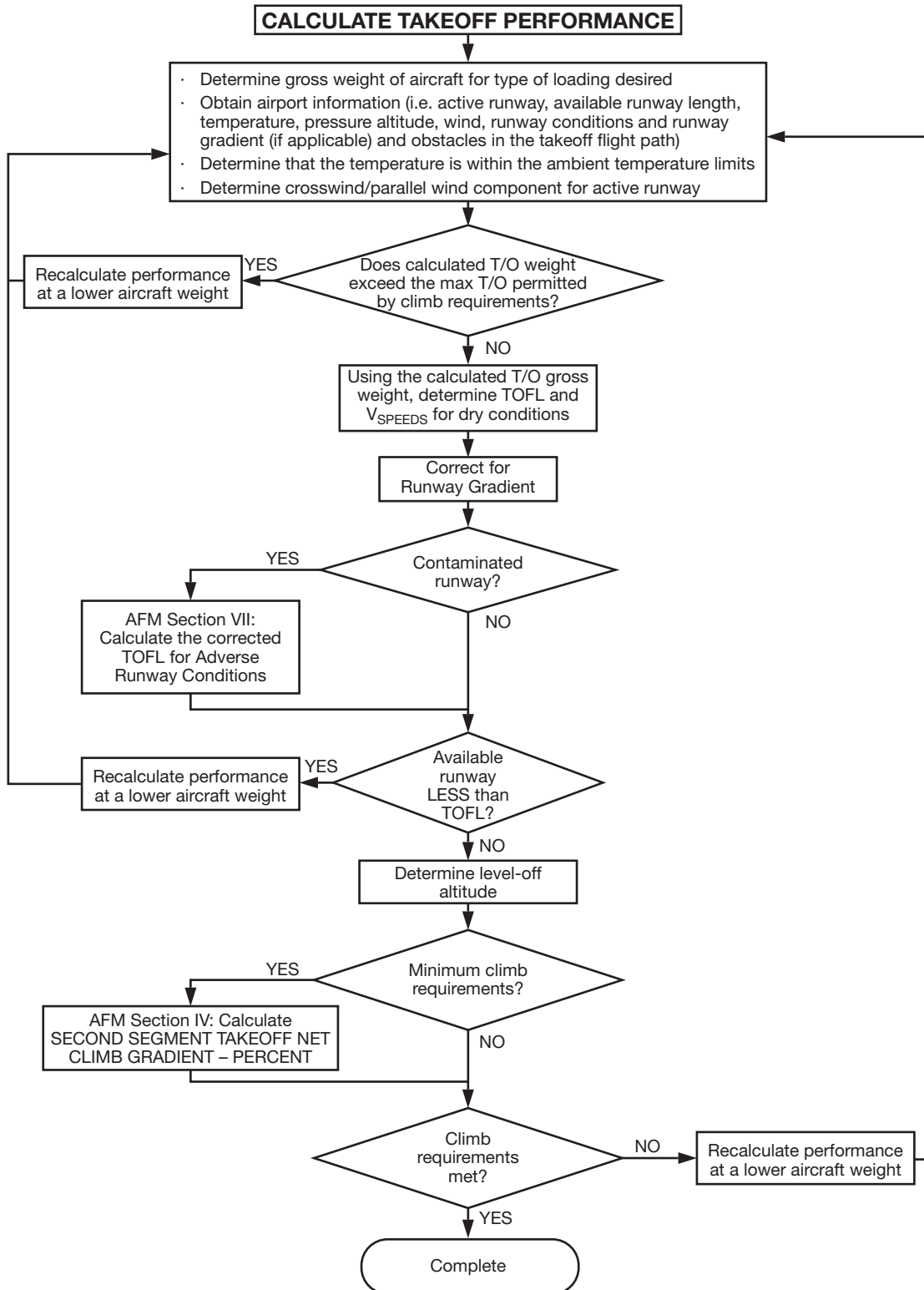


Figure 20-1. Calculation of Takeoff Performance



CITATION 650 SERIES PILOT TRAINING MANUAL

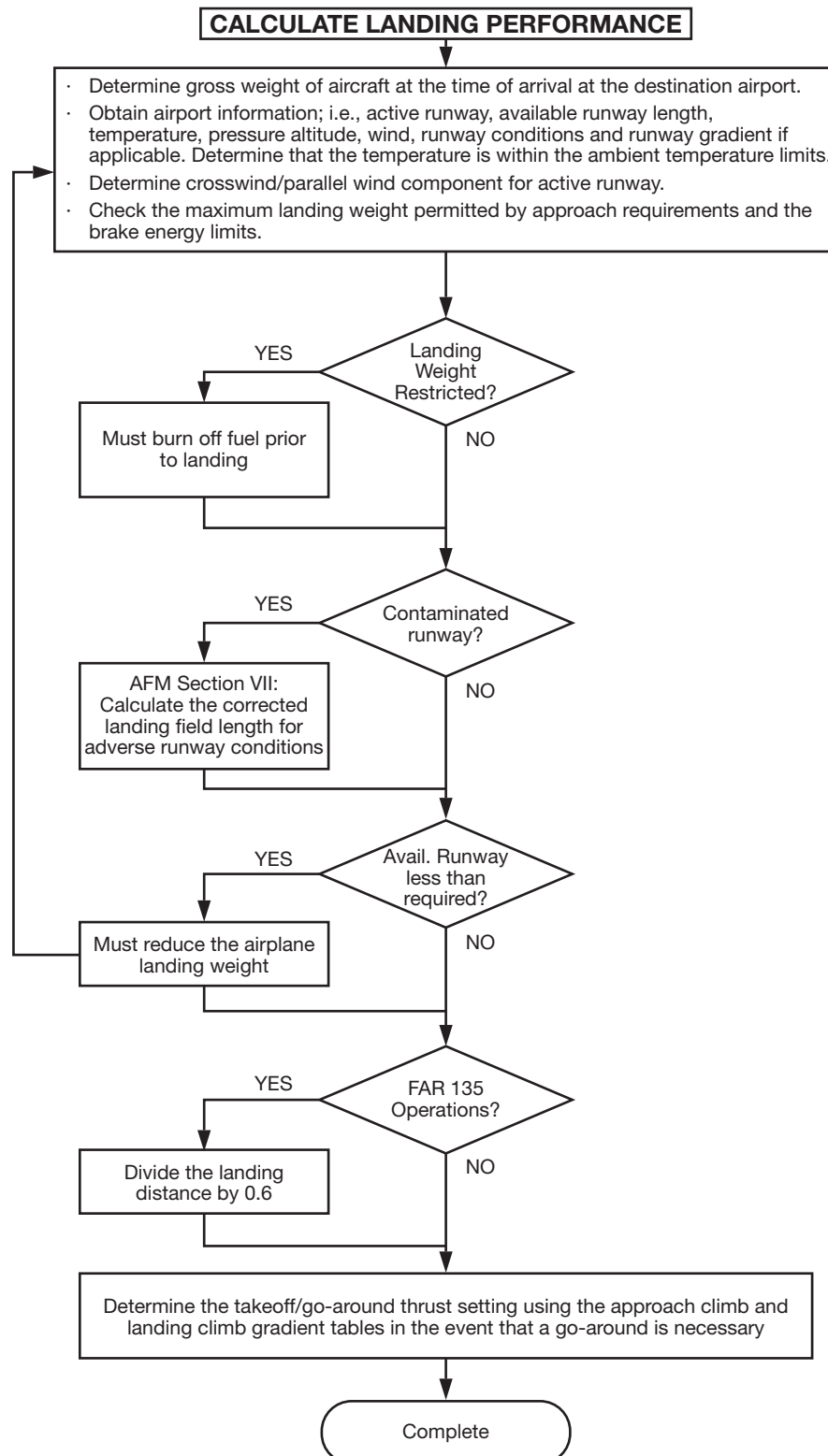


Figure 20-2. Calculation of Landing Performance



CHAPTER 21

CREW RESOURCE MANAGEMENT

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ILLUSTRATIONS

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CHAPTER 21

CREW RESOURCE MANAGEMENT

WHAT IS CREW RESOURCE MANAGEMENT?

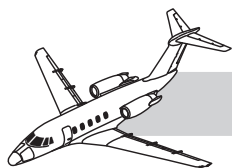
According to the Federal Aviation Administration, Crew Resource Management (CRM) is described as “the effective use of all resources to achieve safe and efficient flight operations.” In practice, CRM is a set of competencies designed to enhance safety and reduce human error. Resources can include, but are not limited to, additional crewmembers, maintenance technicians, flight attendants, air traffic controllers, dispatchers and schedulers, and line service personnel. CRM was not designed to usurp the authority of the pilot in command; rather, it was developed as a means to assist with situational awareness and decision making to increase safety margins and achieve accident- and incident-free flight operations.

Most experts agree that a highly coordinated crew using a standardized set of procedures is more likely to avoid and identify errors. Effective communication and the use of briefing and debriefing are tools that can be used to build the “team concept” and maintain situational awareness. Utilizing a standard set of

callouts provides a means to incorporate CRM. Standardization keeps all crewmembers “in the loop” and provides an opportunity to detect an error early on, before it has an opportunity to build into an accident chain.

Proficiency in CRM requires all crewmembers to have a working knowledge of how to maintain situational awareness, techniques for optimum decision making, desirable leadership and followership characteristics, cross-checking and monitoring techniques, means of fatigue and stress management, and communication.

CRM training is an important part of your FlightSafety training experience. Throughout your training event, your instructor will provide general CRM guidance as well as identify CRM issues, philosophies, and techniques that are specific to the aircraft you fly. To assist with this, the FlightSafety CRM model has been incorporated into this training guide. The model can be used as a guide or a refresher on how to incorporate CRM principles into your day-to-day line operations. This model is not intended to replace a formalized course of CRM instruction, and attendance at a CRM-specific course is highly recommended.



SITUATIONAL AWARENESS

Situational awareness is a fundamental CRM concept. Often described as “knowing what’s going on around you,” the loss of situational awareness is often identified as a causal factor in an incident or accident. Collective situational awareness is a measurement of the total situational awareness among all members involved in the operation. To maintain a high level of collective situational awareness open, timely, and accurate communication is required. In the situational awareness model two-way arrows represent the

two-way communication that must occur between the pilot flying and the pilot monitoring. Each pilot contributes to collective situational awareness.

Circumstances will sometimes present clues that situational awareness is becoming impaired. These “behavioral markers” are listed under clues to identifying loss of situational awareness. As the number of these clues increases, the chance of losing situational awareness increases as well. Maintaining situational awareness requires a constant state of vigilance. Complacency has often been the precursor to a loss of situational awareness (Figure 21-1).

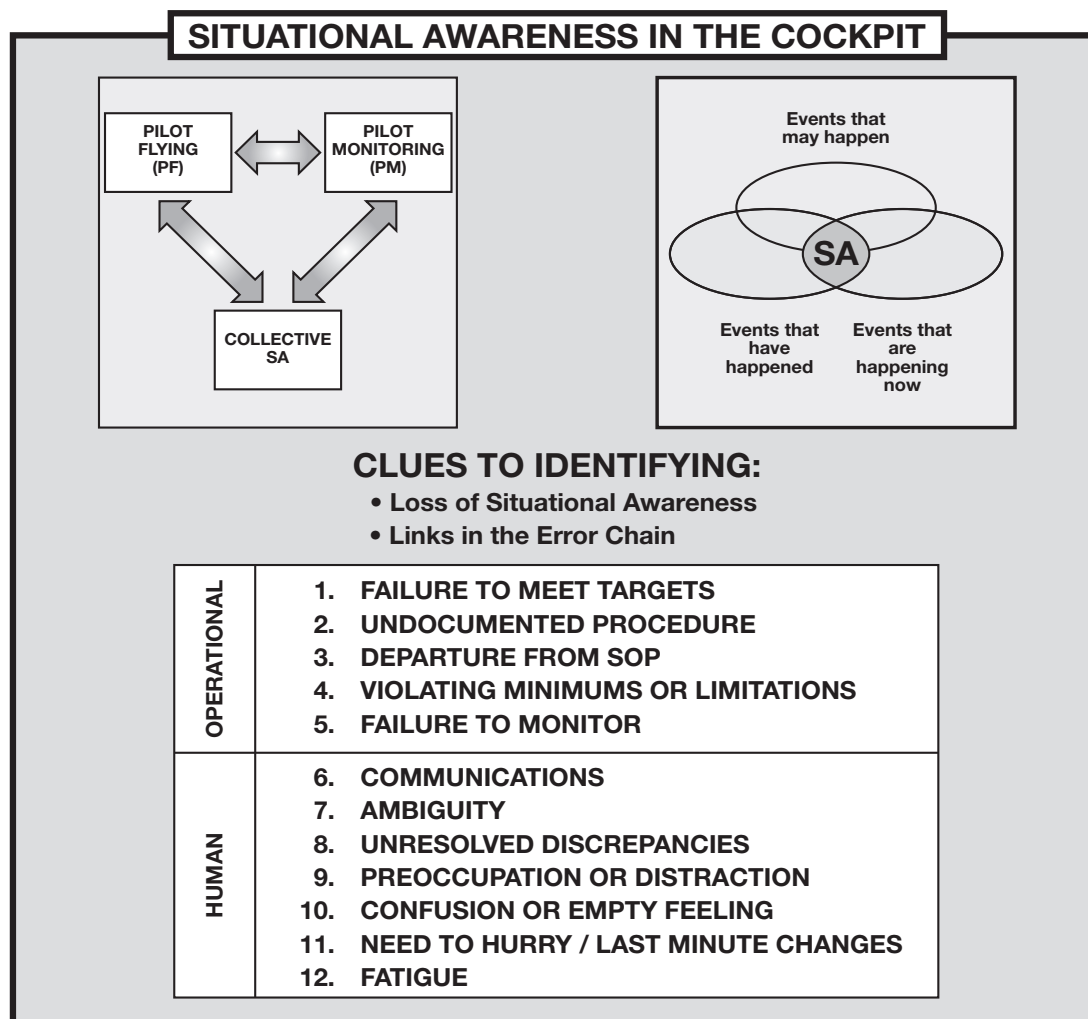
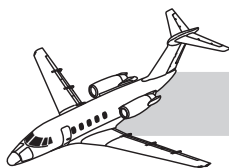


Figure 21-1. Situational Awareness in the Cockpit



COMMAND AND LEADERSHIP

Command and leadership are not synonymous. The status “pilot in command” is designated by an organization. Command responsibility can’t be shared with other crewmembers. Leadership, on the other hand, is a role that can be shared. Effective leadership should focus on “what’s right,” not on “who’s right.”

Leadership styles range from “autocratic” to “laissez-faire.” An autocratic leadership style exercises a high degree of control and allows a low degree of participation from team

members in reaching decisions. A laissez-faire leadership style exercises a low degree of control and allows a high degree of participation from team members. Effective leaders tend to be less extreme, relying on either authoritarian or democratic leadership styles (Figure 21-2).

There is no “ideal” or “best” leadership style. An immediate crisis might require fairly strict leadership, to ensure stability and to reassure other crewmembers, while other situations might be handled more effectively by encouraging crew participation in the decision-making process.

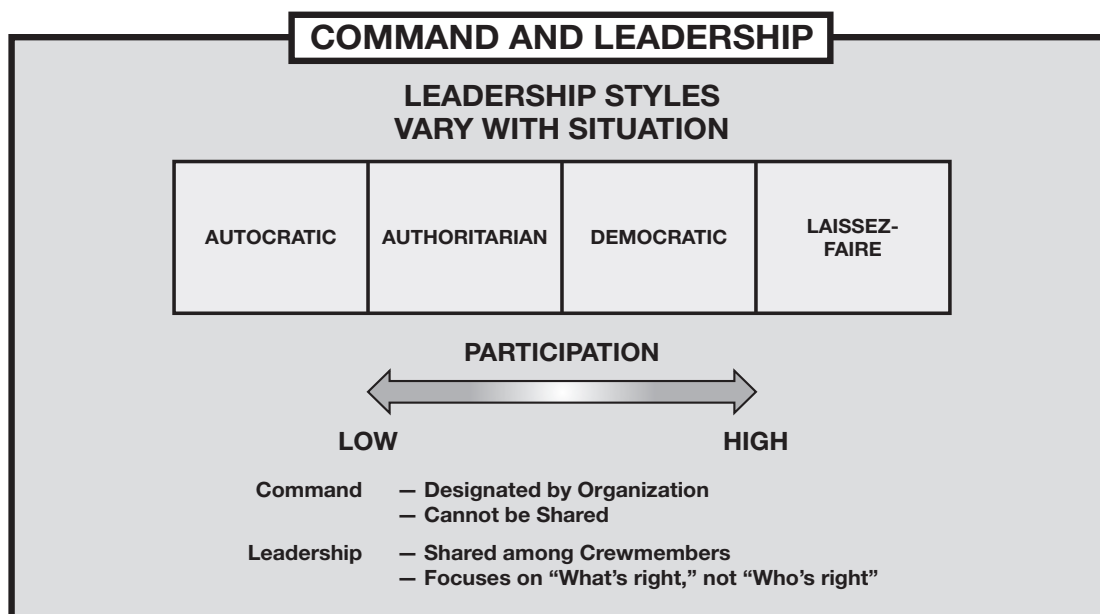


Figure 21-2. Command and Leadership



COMMUNICATION PROCESS

Communication is the most important tool for maintaining situational awareness. Effective communication requires the ability to provide appropriate information, at the appropriate time, to the appropriate person (Figure 21-3). Communication may be verbal (aural) or written. Written communications in the cockpit include symbolic messages and indications that are electronically transmitted and displayed.

As illustrated on the CRM Blue Card, some elements are common to most cockpit communications:

- An event occurs, creating a need to communicate. The event may be a change in the status of some operational goal, such as rate of descent.
- A sender observes the event.
- The sender transmits a message to a receiver, conveying occurrence of the event.
- The receiver transmits feedback to the sender, acknowledging the message.
- The receiver's feedback may include an additional message, confirming the intended corrective action, or instructing the sender to continue monitoring the operational goal.

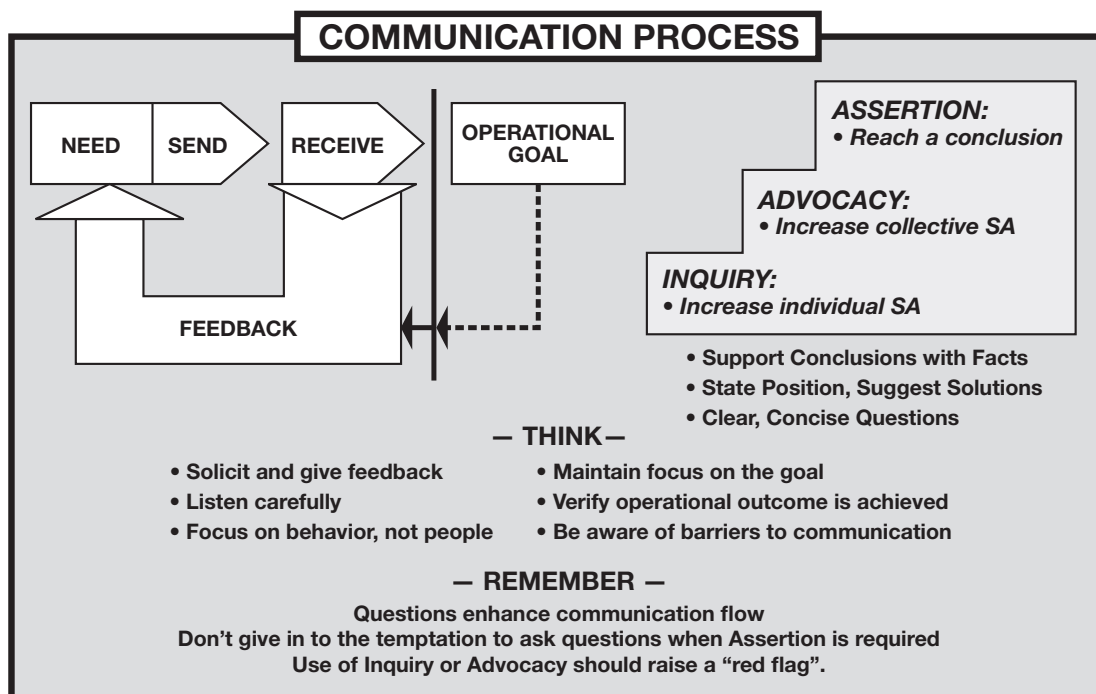
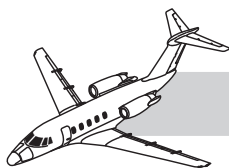


Figure 21-3. Communication Process



Barriers to communication limit our ability to maintain situational awareness.

As illustrated on the Blue Card, internal (or personal) communication barriers can diminish our perception of the need to communicate. An observer who is distracted, for example, may fail to detect a change in the status of an operational goal. Internal barriers can also inhibit a sender's willingness to communicate, or affect a receiver's acceptance and interpretation of a transmitted message.

External communication barriers, such as overcrowded radio frequencies, can interfere with the sender's ability to transmit a message, or with the receiver's ability to transmit feedback. Differences in language or dialect can also become external barriers to communication.

CRM provides three techniques for overcoming communication barriers:

- Inquiry—A technique for increasing your own situational awareness
- Advocacy—A technique for increasing someone else's awareness
- Assertion—A technique for getting your point across

When conflict on the flight deck interferes with communication, it usually originates from one pilot's tendency to make "solo" decisions. Avoid this kind of conflict by focusing your questions and comments on WHAT is right, rather than on WHO is right.

COMMUNICATION TECHNIQUES: INQUIRY, ADVOCACY, AND ASSERTION

Inquiry, advocacy, and assertion can be effectively used in the aviation environment to help solve communication problems.

Each item is a step in the process. The steps provide a metaphor that emphasizes the principle of escalation. In other words, a person must first practice inquiry, then advocacy, then assertion.

A person practicing assertiveness is not trying to be insubordinate or disrespectful; rather, assertion is an expression of the fact that a level of discomfort exists with a particular situation. Assertion is an attempt to seek resolution.

The goal of inquiry is to increase individual situational awareness, the goal of advocacy is to increase collective situational awareness, and the goal of assertion is to reach a conclusion.



DECISION-MAKING PROCESS

Aeronautical decision making (ADM) provides a systematic approach to risk assessment. It is a tool you can use to select the best response for a given set of circumstances. FlightSafety recommends the decision-making process illustrated on the second page of the Blue Card (Figure 21-4). This continuous-loop process includes eight steps:

1. Recognize the need for a decision.
2. Identify the problem and define it in terms of time and risk.
3. Collect facts.
4. Identify alternative responses to the need.
5. Weigh the impact of each alternative response.
6. Select a response.
7. Implement that response.
8. Evaluate the effects of your response.

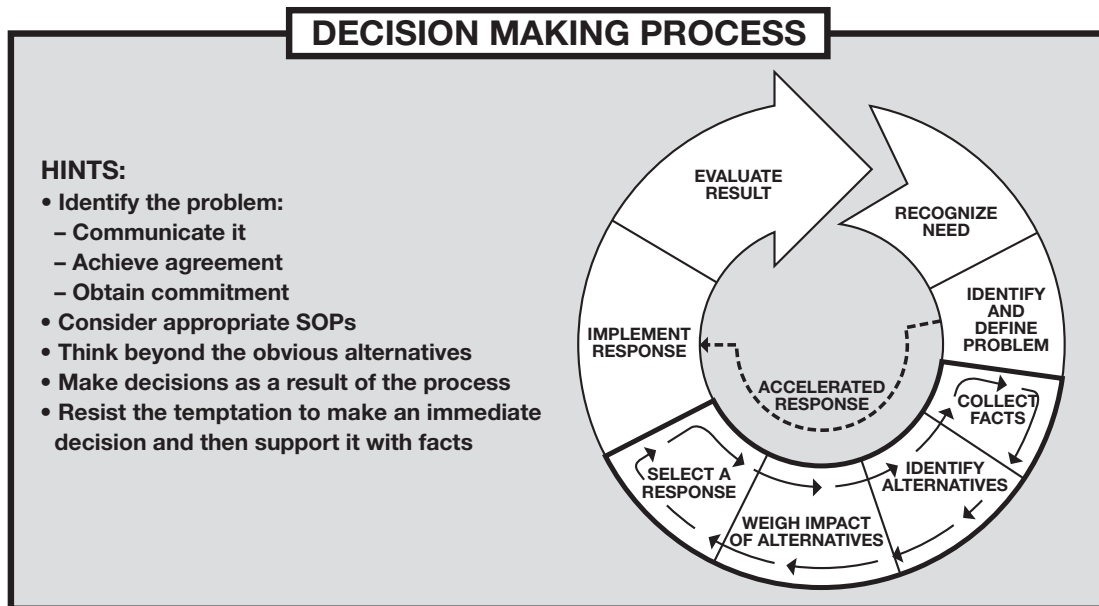
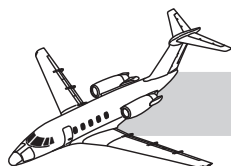


Figure 21-4. Decision-Making Process



The information normally contained in this chapter can be found
in the Citation 650 series electronic Walkaround Program.



APPENDIX

ANSWERS TO QUESTIONS

CHAPTER 2

1. C
2. B
3. D
4. A
5. B
6. D
7. C
8. D
9. A
10. B
11. C
12. A
13. D
14. C
15. A

CHAPTER 3

1. A
2. D
3. D
4. D

CHAPTER 4

1. C
2. D
3. A

CHAPTER 5

1. C
2. B
3. D
4. A
5. A
6. D
7. A
8. B
9. C
10. B

CHAPTER 6

1. D
2. B
3. A
4. D

CHAPTER 7

1. D
2. A
3. B
4. A
5. D
6. B
7. B
8. C
9. C
10. C
11. B
12. C
13. A

CHAPTER 8

1. A
2. C
3. B
4. D
5. B
6. B

CHAPTER 9

1. B
2. B

CHAPTER 10

1. A
2. B
3. D
4. A
5. C
6. B
7. A
8. C
9. D
10. B
11. C
12. D

CHAPTER 11

1. A
2. B
3. D
4. B
5. D
6. A
7. D
8. A
9. A
10. D

CHAPTER 12

1. B
2. D
3. A
4. C
5. B

CHAPTER 13

1. A
2. C
3. B
4. C
5. A
6. D
7. D
8. C

CHAPTER 14

1. B
2. D
3. A
4. B
5. C
6. D
7. B
8. D
9. A
10. C
11. D
12. D
13. C
14. A

CHAPTER 15

1. A
2. B
3. B
4. D
5. A
6. D
7. B
8. B
9. D
10. A
11. A
12. A
13. C
14. B

CHAPTER 17

1. C
2. D
3. B
4. A
5. C
6. C
7. D
8. C



Refer to Chapter 4—“Master Warning System” for the information normally contained in this chapter.

